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PROJECT APOLLO
MISSION OPERATIONS ANALYSIS
FOR A
TYPICAL LUNAR LANDING MISSION
(U)

28 January 1963

4.5.9.4

Prepared by

Operations Analysis
Aerospace Science

CLASSIFICATION CHANGE
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FOREWORD

The Apollo Lunar Landing Mission described and analyzed in this report is a hypothetical mission. The events and conditions described are to be considered only as representative of those which may occur in an actual mission. Other representative missions have been or will be generated to indicate various requirements, such as maximum duration, or to establish perspective as to the envelope of acceptable missions. The mission presented herein, which closely agrees in concept with one independently generated in the same time period by the NASA Manned Spacecraft Center, is to serve as a realistic basis for evaluation of requirements and for planning an early lunar-landing mission.

This document supplements the previous edition of SID 62-379. Where earlier data on the lunar landing mission appear in conflict with the data in this (SID 62-379-1) document, the latter will generally prevail.

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INTRODUCTION

Insight into a number of aspects of mission operations can be gained only from the perspective of a totally defined mission which shows the order of events and the interplay between crew functions, system functions and ground operations as related to real time.

A specified mission, therefore, was developed to serve as a model for future studies and to generally formulate, as clearly as possible, a description of mission elements and their organization for the benefit of interested groups within Project Apollo.

A lunar landing type mission was chosen for this model because it introduces a majority of considerations. This mission is neither maximal nor minimal in regards to duration or objectives. This is a "normal event" mission and does not deal with contingencies and abort situations. The mission is herein referred to as typical, or one in which each mission phase is representative of a realistic flight situation. The typical mission was developed around operational characteristics - such as landing in daylight and on land, which are commonly thought to be ideal specifications, although such ideas are currently under investigation.

With this typical mission as a framework, the document presents answers to a number of pertinent questions concerning the integrated operation, including: <u>What</u> systems (Spacecraft and/or GOSS) are involved in particular events; <u>Where</u> do various events occur geometrically in relationship to the mission; <u>When</u> do the events occur; <u>How</u> does the space vehicle configuration change during the mission.

Section I, Mission Description, introduces the typical lunar landing mission which is to analyzed. A specification of the mission objectives is followed by a listing of desirable operational characteristics. Such characteristics are reflected in presentation of the explicit mission trajectory which has been selected. An earth trace diagram and a mission time history are also included.

Section II, Time-Line Summary, is a chronological listing of all major mission events (by phase) during the mission. These events pertain to the Apollo spacecraft or particular booster stages of the C-5 launch vehicle.

Section III, Mission Phase Analysis, is a detailed analysis of spacecraft system activity during each of the 12 mission phases. In addition to geometry considerations and a listing of mission events and requirements for each mission phase, the pertinent functions of each spacecraft system have been documented in detail in APPENDIX F, and are plotted along a time scale for that particular phase.

A basic premise in this mission analysis is to emphasize the operation of on-board systems, whether or not they will ultimately have a primary or backup role in an actual mission. Guidance and navigation parameters for example, will probably be determined independently by GOSS and the Spacecraft. This document, however, is not presently concerned whether the on-board spacecraft or GOSS determination shall prevail.

As supporting information to the mission analysis, particular aspects of a lunar landing mission are covered in more detail in a series of appendices.

SECTION I

MISSION DESCRIPTION

Mission Objectives

The ultimate objective of Project Apollo is to land men on the moon for limited observation and exploration in the vicinity of the landing area, and subsequent safe return to earth. This objective will climax a series of earth orbital, circumlunar and lunar orbital missions. Although each of these missions will have specified objectives, they will be flown primarily for state-of-the-art advancement and qualification of systems for the ultimate lunar landing mission. Unique objectives of the lunar landing mission include:

(1) LEM lunar landing

(2) Lunar surface exploration

(3) Mission Verification

(4) Evaluation of crew reaction on lunar surface

(5) One Man Crew Command Module Operation in Lunar Orbit

(6) Two-Manned Lunar Launch

(7) Lunar Orbit Rendezvous

Operational Characteristics

A variation in operational characteristics will be reflected in trajectory design for each Apollo mission. The typical lunar landing mission analyzed in this document is based on a trajectory having the following specifications:

(1) Mission flight date - 1967.

(2) Planned earth landing site-vicinity of San Antonio.

(3) Lunar landing site-within 0 to +10 degrees N. latitude and 20 to 40 degrees W. longitude. (approximate impact area for Ranger & Surveyor)

(4) Lunar lighting conditions at landing.- daylight (high-moon conditions to be avoided if possible)

(5) Earth lighting conditions at landing near San Antonio.- daylight, at least 2 hours before dusk. The minimum range contingency landing area in the Pacific should also be in daylight.

(6) Entry trajectory plane to San Antonio - inclined between 29 and 34 degrees to the equator for favorable GOSS coverage.

(7) Launch from Cape Canaveral should be made in daylight, no later then 4 hours before dusk.

(8) Launch azimuth should be within 75 to 95 degrees.

(9) Translunar injection should occur on the second earth parking orbit over the Pacific, i.e. the "long-coast" injection area.

(10) The spacecraft should make at least 2 and no more than 4 passes over the lunar landing area before LEM separation.

(11) The LEM should make at least 1 and no more then 2 passes over the lunar landing area before initiating descent for landing.

(12) Lunar stay time - 4 to 8 hours.

(13) Rendezvous and Docking time allowance - 0 to 3 hours.

(14) The spacecraft should make at least one complete lunar orbit after LEM jettison to allow time for transearth injection preparations.

TRAJECTORY CHARACTERISTICS

In accordance with the above operational specification, a representative (typical) lunar landing mission was synthesized using existing trajectory data, ephemeris data defining the motion of the sun and moon, and certain geometric relationships and dynamic considerations required to yield a continuous, though approximate, mission profile. The circumlunar, free-return trajectory used in the analysis was integrated in a simplified model having the moon in a circular orbit at a mean distance from the earth. Impulse velocity increments were assumed for all powered flight phases except the boost to earth parking orbit and the translunar injection. An exact launch time is not specified because of the approximate nature of the trajectory analysis. Continuity in time is maintained, however, throughout the mission.

Earth Vicinity & Translunar

Figure 1 is a schematic summary of the trajectory characteristics for the translunar portion of the lunar landing mission. Launch occurs at Cape Canaveral on August 14, 1967 about 7.7 hours before dusk. The launch azimuth is 78.275 degrees. The powered flight phase from launch to parking orbit injection at an altitude of 100 n. miles requires 706 seconds and a central in-plane angle of 23.5 degrees.

The parking orbit cost central angle is 671.15 degrees and the coast time is 163.7 minutes. The powered flight phase from parking orbit to translunar injection requires 312 seconds and has a central angle of 23.5 degrees. Translunar injection occurs over the Pacific (North

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latitude 29.6 degrees, 120.2 degrees west longitude about 7.6 hours prior to dusk. The injection inertial velocity is 35,860 fps. The translunar trajectory plane defined by the radius and velocity vectors at injections is inclined 30.6 degrees to the Earth equator plane and 20 degrees to the lunar orbit plane.

The coast time from translunar injection to perilune arrival is 64.84 hours. The moon at this time is 280.98 degrees from the lunar orbit plane ascending node on the Earth equator and has a declination of -27.35 degrees.

<u>Lunar Vicinity</u>

Figure 2 shows the lunar vicinity trajectory. Injection into an 80 n. mile altitude circular orbit occurs at perilune, near the extended Earth-Moon line of centers and in darkness. The maneuver is co-planer (ΔV = 3143 fps) since the translunar injection velocity was selected such that the plane of the incoming lunar conic contained the landing site (selenographic latitude 6 degrees North, longitude 30.30 degrees west - toward the leading edge). The orbital period is 2.0438 hours. The circular orbit plane is inclined 8.83 degrees to the lunar orbit plane and the ascending node on the lunar orbit plane is 156.22 degrees counter-clockwise from the extended Earth-Moon line at the time of perilune arrival.

The spacecraft makes two passes over the landing site prior to LEM separation, at which time it has been in the circular orbit for 288.7 minutes and traveled through 847.6 degrees of central angle. The LEM separates at a central angle of 94 degrees prior to the third pass over the landing site. The LEM separation ΔV is 373 fps.

The LEM makes one pass over the landing site, in the equal period orbit and initiates retro thrust near perilune the next time around. The central angle of the equal period orbit is 454 degrees and the elapsed time in orbit is 154.7 minutes. Perilune velocity is 5671 fps.

At the time of LEM landing on the lunar surface, it is a lunar morning, the phase of the moon being midway between first quarter and full Moon. The elevation of the Sun is 21 to 23 degrees above the local horizon plane during the 5.87 hours of surface stay time. Figure 4 is a lunar relief map of the landing site.

At lunar departure the LEM boosts to 50,000 ft. where it is injected into an ascent ellipse having a perilune velocity of 5,580 fps. Rendezvous occurs at apolune of the ascent ellipse about 58 minutes after injection. The ΔV at apolune is 97 fps. Transearth injection occurs in darkness near the extended Earth-Moon line during the second orbit, 3.89 hours (685 degrees central angle) after rendezvous. The transearth injection ΔV is 3405 fps.

Transearth & Earth Vicinity

Figure 3 shows the transearth coast trajectory. Transearth coast time from injection to entry is 82.41 hours. The geocentric trajectory plane inclination to the Earth equator plane at entry is 34 degrees. Entry (h = 400,000 ft) occurs 16.92 degrees North Latitude and 131.9 degrees West Longitude, about 3.1 hours after dawn. The entry range to San Antonio is 1817 n.miles. Landing occurs about 7.3 hours prior to dusk, after a total mission duration of 7 days + 0.9 hours.

Two additional figures are included as part of the lunar landing mission description. Figure 5 presents for the entire mission the earth surface trace of an imaginary line between the center of the earth and the moving spacecraft. Table 1 presents the mission time history and vehicle configuration for each mission phase.

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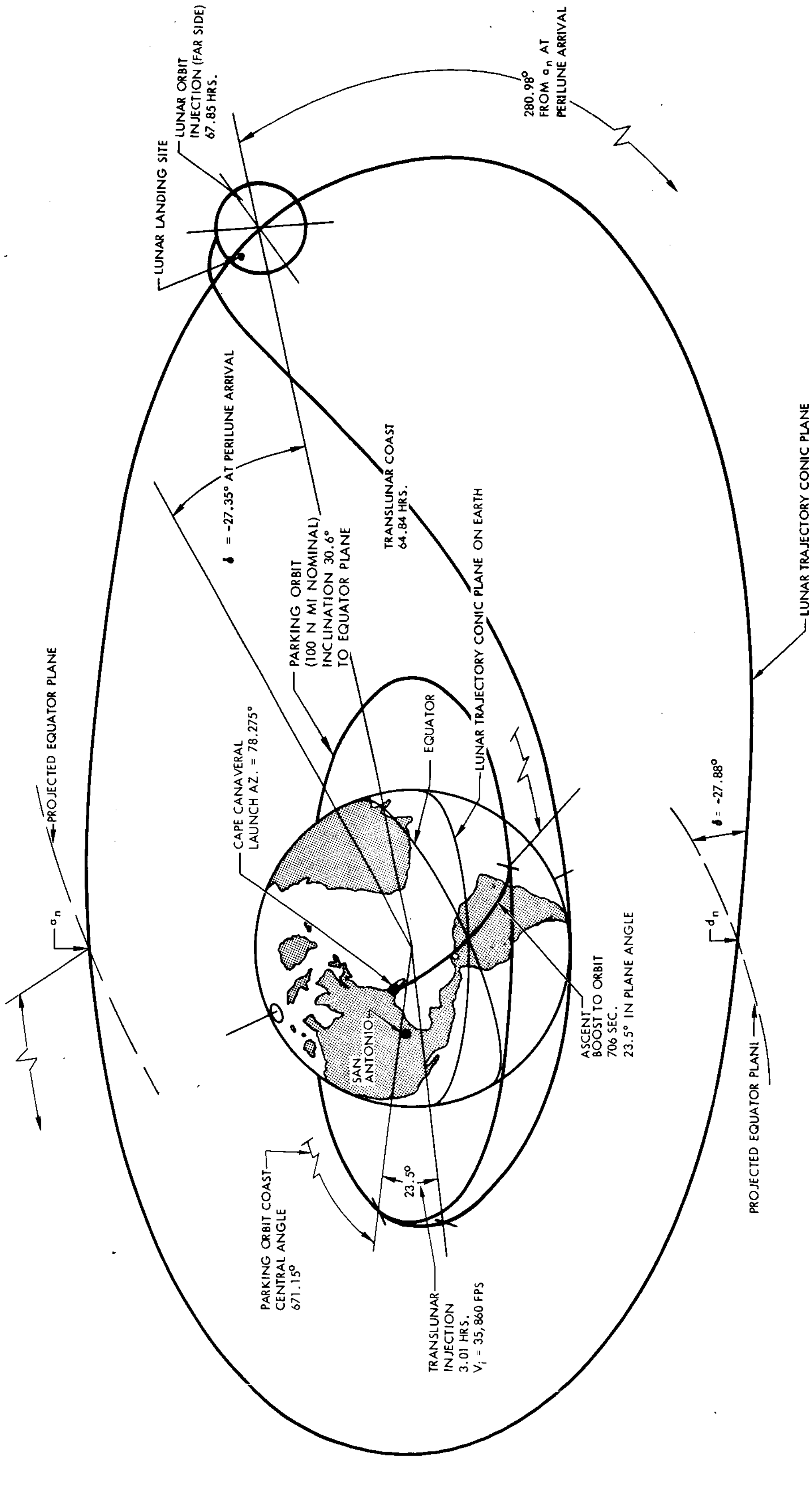

Figure 1. Trajectory Characteristics - Translunar

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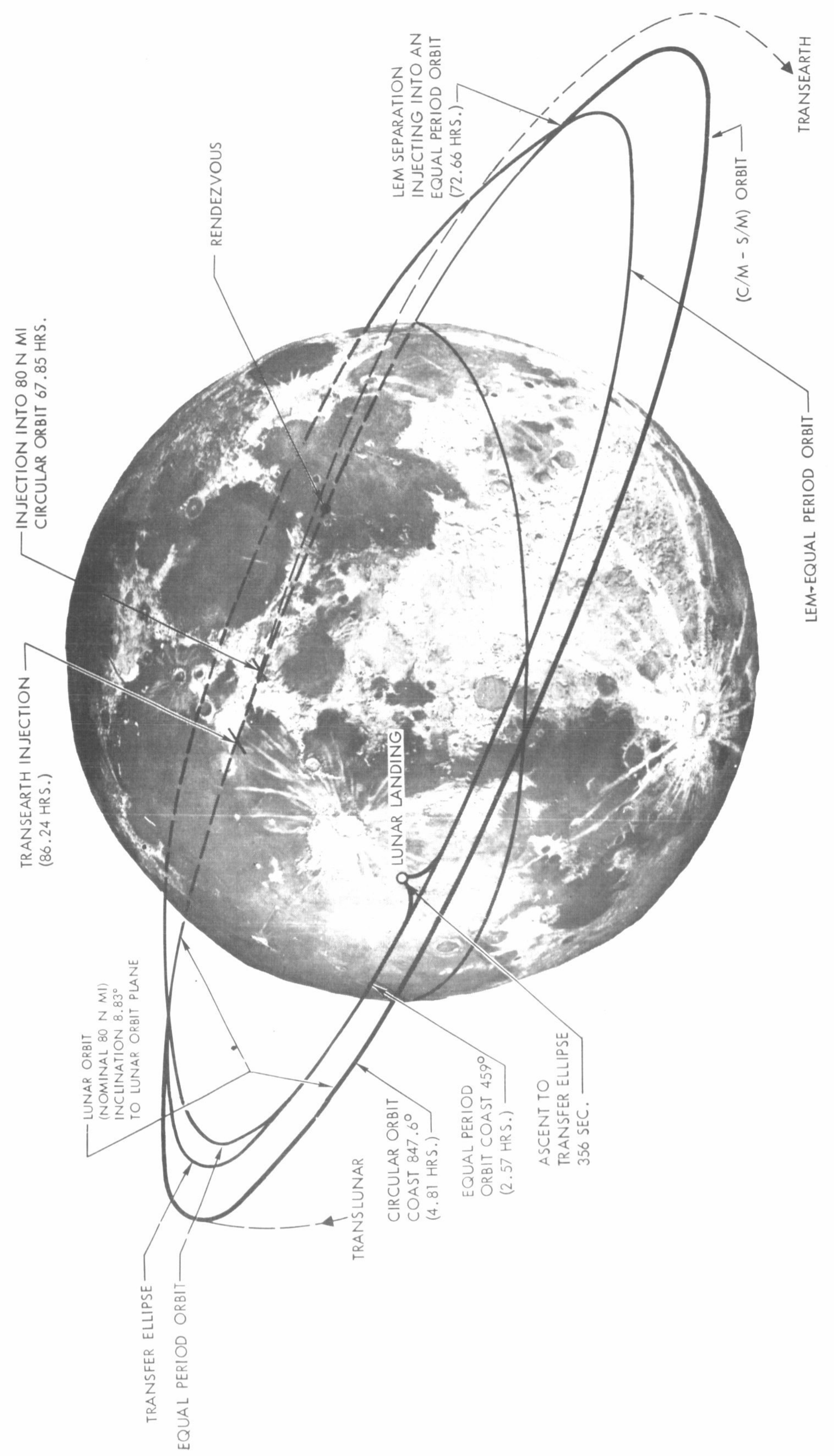

Figure 2. Trajectory Characteristics - Lunar Vicinity

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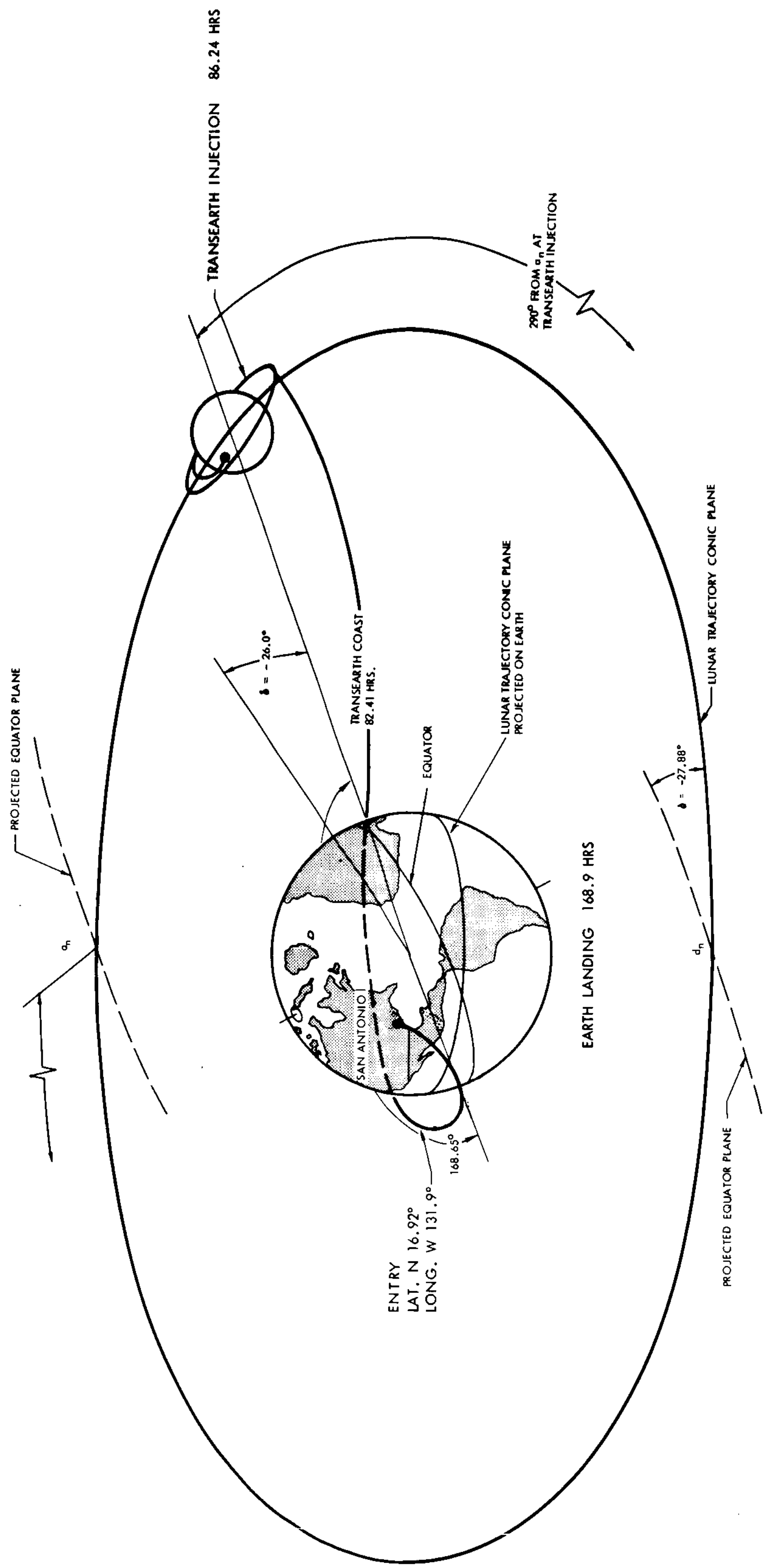

Figure 3. Trajectory Characteristics - Transearth

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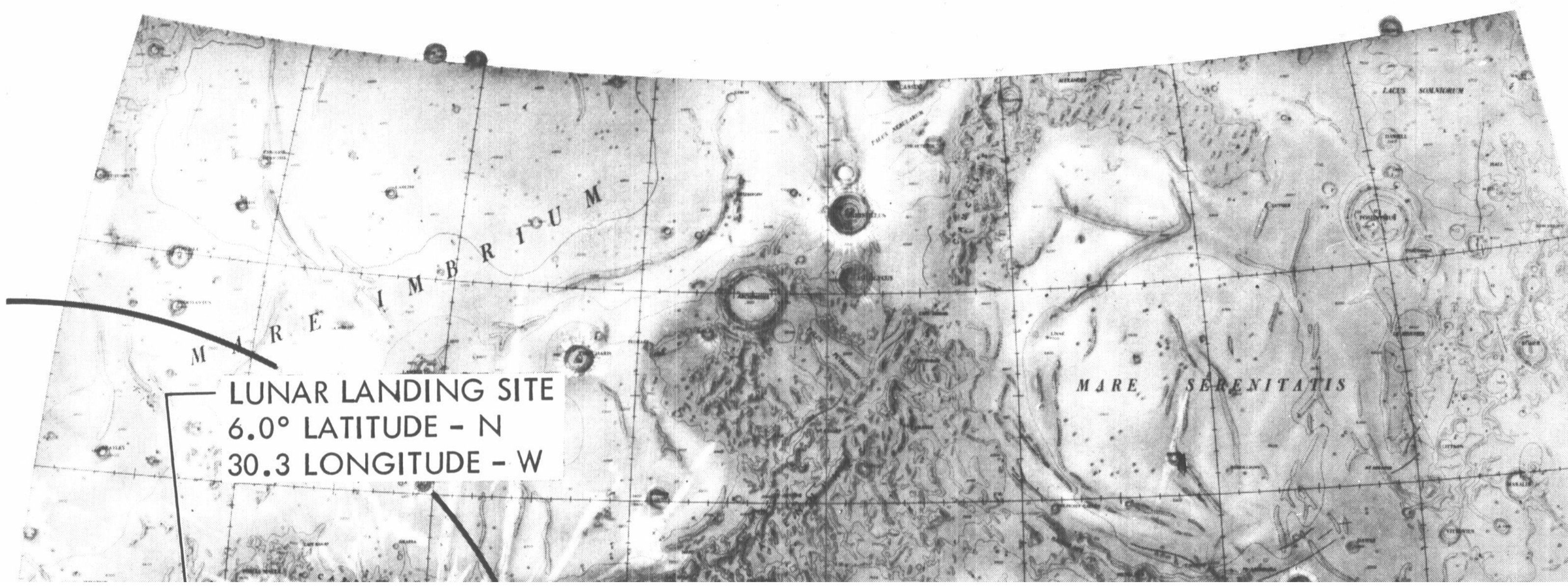
MARE NECTARIS–MARE IMBRIUM



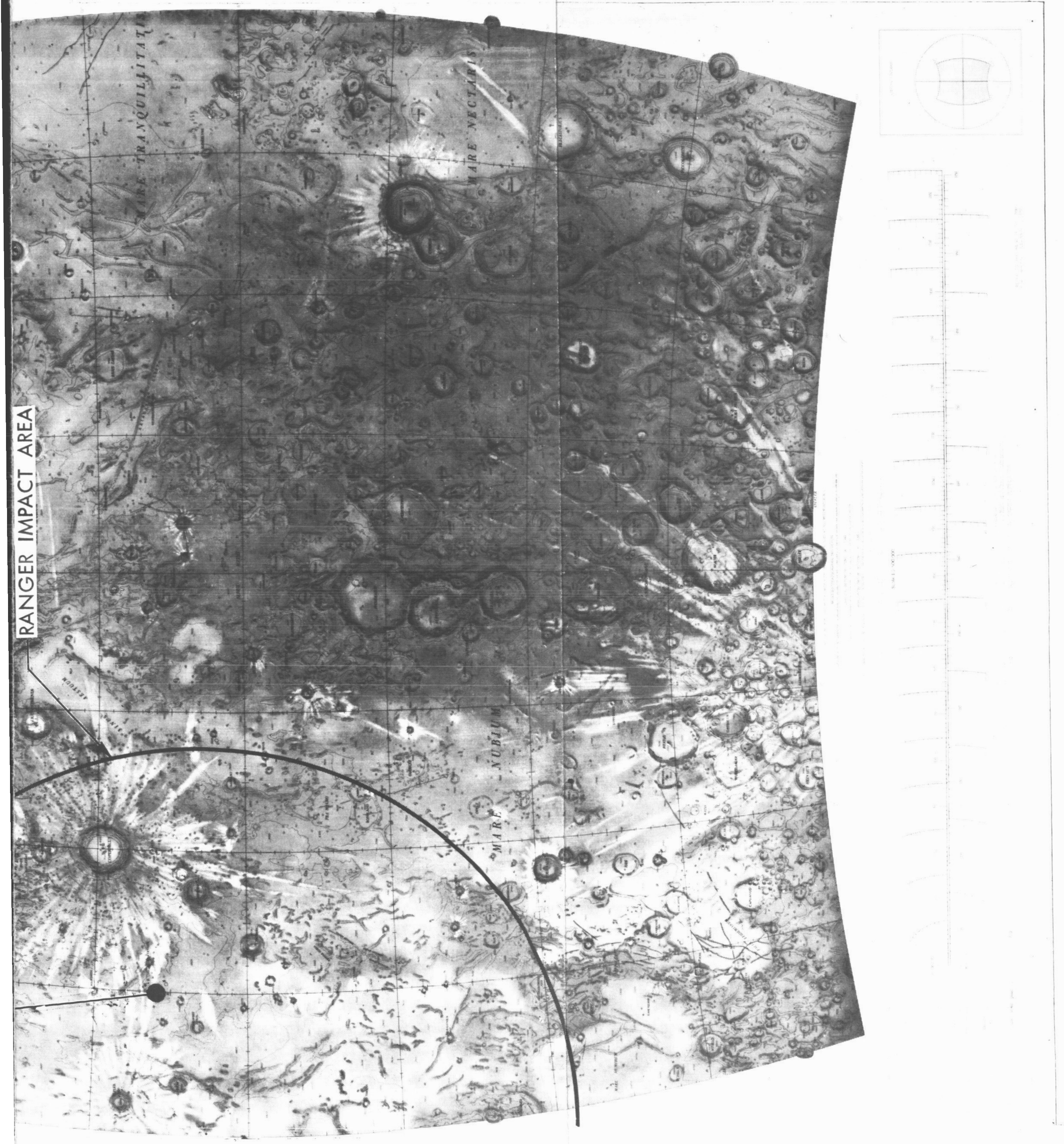


Figure 4. Lunar Landing Site (Mare Nectaris-AMS)

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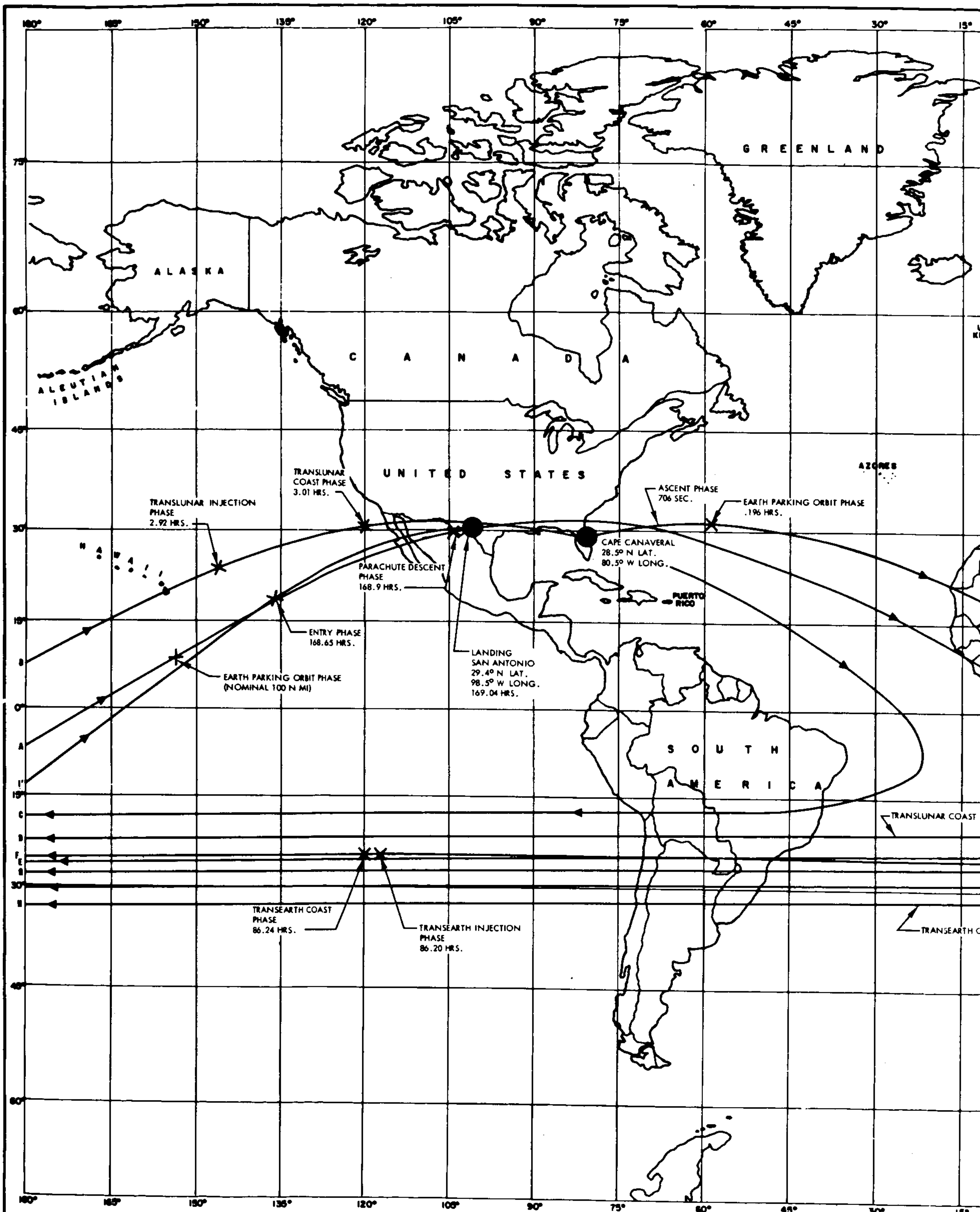

GREENLAND
ALASKA
CANADA
ALEUTIAN ISLANDS
UNITED STATES
AZORES
HAWAII
PUERTO RICO
SOUTH AMERICA
TRANSLUNAR COAST PHASE 3.01 HRS.
TRANSLUNAR INJECTION PHASE 2.92 HRS.
ASCENT PHASE 706 SEC.
EARTH PARKING ORBIT PHASE .196 HRS.
CAPE CANAVERAL 28.5° N LAT. 80.5° W LONG.
PARACHUTE DESCENT PHASE 168.9 HRS.
ENTRY PHASE 168.65 HRS.
LANDING SAN ANTONIO 29.4° N LAT. 98.5° W LONG. 169.04 HRS.
EARTH PARKING ORBIT PHASE (NOMINAL 100 N MI)
TRANSLUNAR COAST
TRANSEARTH COAST PHASE 86.24 HRS.
TRANSEARTH INJECTION PHASE 86.20 HRS.

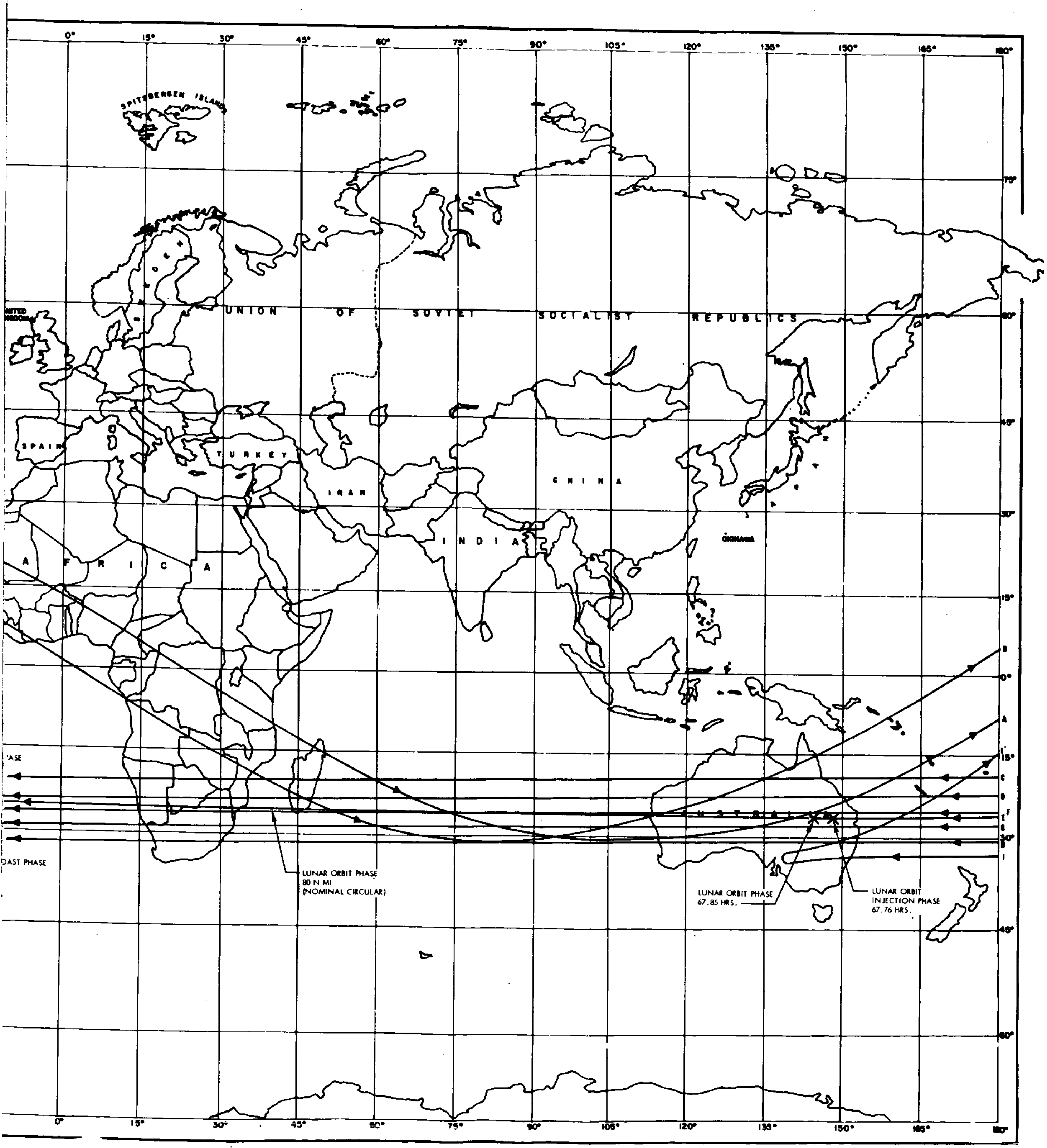

Figure 5. Mission Trajectory Earth Trace

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TABLE 1

LUNAR LANDING MISSION TIME SUMMARY

Space Vehicle Configuration	Mission Phase	Mission Phase Time	Mission Accumulated Time (hrs)
SIC + SII + SIVB + CM + SM + LEM	Launch		0.0
	Boost to Orbit	706.0 sec	0.20
	Earth Parking Orbit Coast	163.7 min	2.42
	Boost to Translunar Injection	312.0 sec	3.01
	Translunar Coast to Perilune	64.75 hrs	67.76
CM + SM + LEM	Lunar Parking Orbit Injection	320.0 sec	67.85
	Lunar Parking Orbit Coast	288.7 min	72.66
LEM	EPO Injection	35.3 sec	72.67
	EPO Coast	154.7 min	75.25
	Perilune Retro (50 K to 1K)	336.0 sec	75.34
	Hover, Translation and Touch-down (1K to Surface)	127.0 sec	75.37
	Surface Stay Time	5.87 hrs	81.24
	Ascent to Transfer Ellipse	356.0 sec	81.34
	Ascent Ellipse Coast	58.1 min	82.31
	Parking Orbit Injection and Rendezvous	5.26 sec	82.31
CM + SM	Lunar Parking Orbit Coast	3.89 hrs	86.20
	Transearth Injection	127.5 sec	86.24
	Transearth Coast	82.41 hrs	168.65
CM	Entry (Point of Deployment of Pilot Chute)	15.0 min	168.90
CM	Parachute Descent	509.0 sec	169.04

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SECTION II

TIME-LINE SUMMARY

This part presents a summary of the time-line mission events during a typical lunar landing mission. The trajectory data which constitutes a framework for this time-line delineation is found in the preceding part of the document. Figure 6 is a chronological listing of mission events during each phase of the mission. In addition to beginning and ending times for each phase as part of total elapsed time during the mission, a non-linear time scale for each mission phase is included. Time of occurrence of mission events is denoted along the time scales for each phase, and the approximate duration for many of the mission events is indicated in parenthesis. These mission events pertain to the spacecraft and/or launch vehicle as an entity and not to individual spacecraft systems which are considered in subsequent parts of the document.

Approximately 72 hours after lift-off and while in lunar orbit, the LEM is separated from the C/M and S/M. From this point until the LEM performs a rendezvous and docking maneuver with the C/M-S/M approximately 10 hours later, the time-line activity is presented in two columns as seen on page 2 of Figure 6. The left column represents the supporting activity of the C/M-S/M (1 crewman) in lunar orbit, while the right column represents the lunar landing activity of the LEM (2 crewmen).

Figure 7 also summarizes the major mission events during the entire mission. Space vehicle activity during each phase of the mission is displayed pictorially to complement the operational sequence of activity.

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MISSION TIME	PHASE	PHASE TIME	MISSION EVENT
T = 0	ASCENT	0	S-IC PROPULSION SYSTEM IGNITION (SPACE VEHICLE LIFT-OFF)
		12 SEC	SPACE VEHICLE PITCHOVER
		143 SEC	S-IC 1 ENGINE CUT-OFF
		148 SEC	S-IC 4 ENGINE CUT-OFF
			S-IC SEPARATION
			S-II ULLAGE ROCKET IGNITION
		150 SEC	S-II PROPULSION SYSTEM IGNITION
		155 SEC	LAUNCH ESCAPE SYSTEM JETTISON
		544 SEC	S-II 5 ENGINE CUT-OFF
			S-II SEPARATION
			S-IVB ULLAGE ROCKET IGNITION (3 SEC)
		547 SEC	S-IVB PROPULSION SYSTEM IGNITION
T = 0.196 HR		ET 706 SEC	S-IVB ENGINE CUT-OFF
	EARTH PARKING ORBIT	0	POST INJECTION CHECK (10 MIN)
		10 MIN	SYSTEMS CHECKOUT (15 MIN)
		25 MIN	EARTH ORBIT PARAMETERS CHECK (75 MIN)
		100 MIN	PREPARATION FOR TRANSLUNAR INJECTION (30 MIN)
		135 MIN	TRANSLUNAR INJECTION COUNTDOWN
T = 2.92 HR		ET 163.7 MIN	S-IVB ULLAGE ROCKET IGNITION (4 SEC)
	TRANSLUNAR INJECTION	0	
		2 SEC	S-IVB PROPULSION SYSTEM IGNITION
T = 3.01 HR		ET 314 SEC	S-IVB ENGINE CUT-OFF
	TRANSLUNAR COAST	0	POST INJECTION CHECK (15 MIN)
		0.50 HR	LEM TRANSPOSITION (15 MIN) ADAPTER SEPARATION S-IVB SEPARATION
		1.00 HR	PREPARATION FOR COAST (20 MIN)
		1.31 HR	SYSTEMS CHECKOUT (50 MIN) NAVIGATION FOR 1ST MIDCOURSE CORRECTION (5 HR., 10 SIGHTINGS)
		6.31 HR	SERVICE PROPULSION SYSTEM IGNITION
		7.00 HR	NAVIGATION FOR 2ND MIDCOURSE CORRECTION (5 HR., 10 SIGHTINGS)
		12.00 HR	SERVICE PROPULSION SYSTEM IGNITION
		17.00 HR	NAVIGATION FOR 3RD MIDCOURSE CORRECTION (30 HR., 10 SIGHTINGS)
		47.00 HR	SERVICE PROPULSION SYSTEM IGNITION
		52.00 HR	NAVIGATION FOR 4TH MIDCOURSE CORRECTION (10 HR., 10 SIGHTINGS)
		62.00 HR	SERVICE PROPULSION SYSTEM IGNITION
		63.75 HR	PREPARATION FOR LUNAR ORBIT INJECTION (1 HR)
T = 67.76 HR		ET 64.75 HR	S/M RCS ULLAGE ACCELERATION
	LUNAR ORBIT INJECTION	0	
		2 SEC	SERVICE PROPULSION SYSTEM IGNITION
T = 67.85 HR		ET 322 SEC	SERVICE PROPULSION SYSTEM CUT-OFF

ET = ELAPSED TIME

Figure 6. Lunar Landing Mission Time Line Summary (Sheet 1 of 2)

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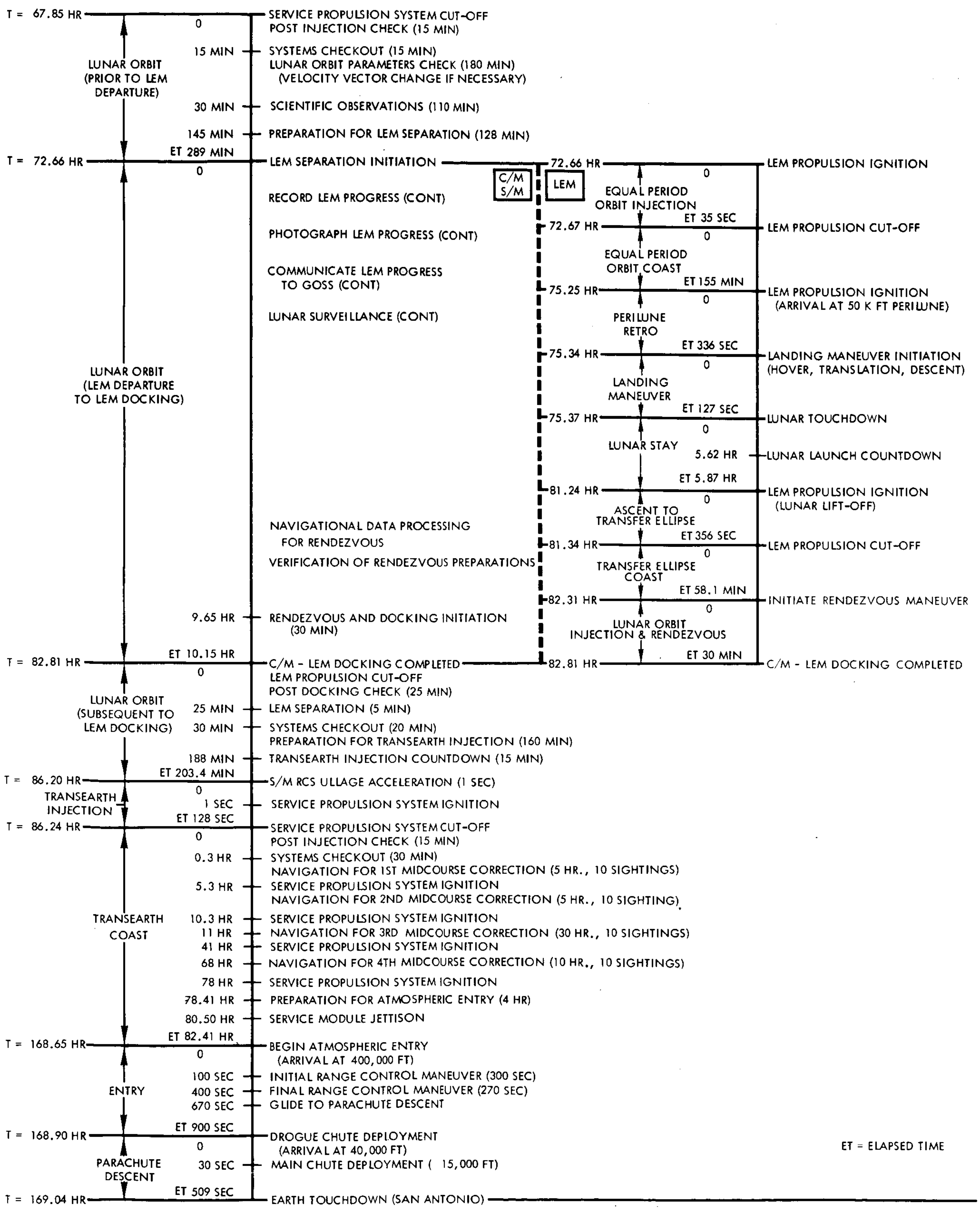

Figure 6. Lunar Landing Mission Time Line Summary (Sheet 2 of 2)

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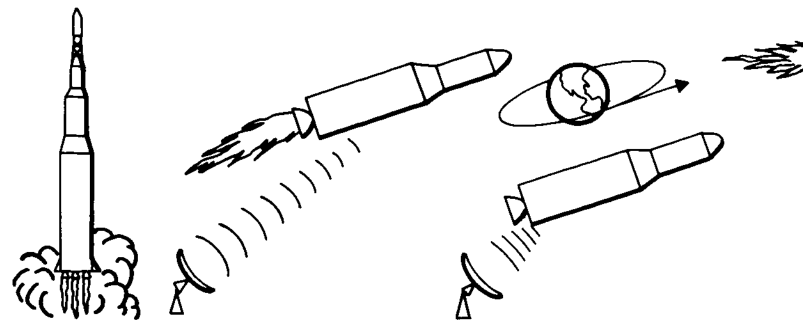
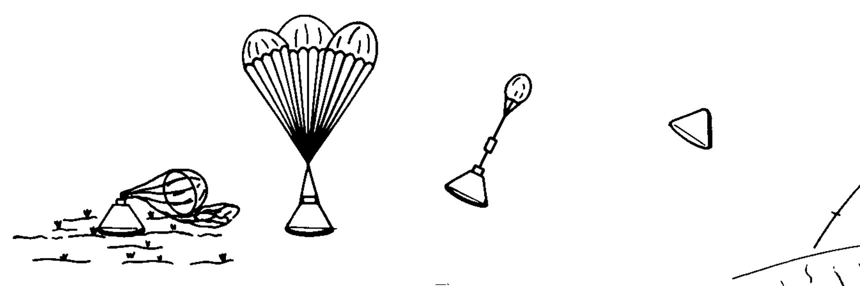


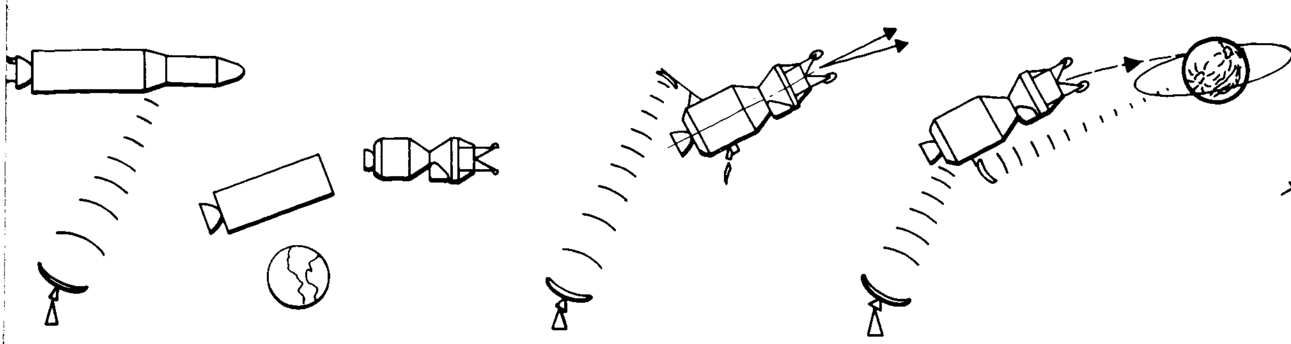
FIGURE / PHASE	ASCENT		EARTH PARKING ORBIT		TRANSL INJEC
PRE LAUNCH	LAUNCH	BOOST	ORBITAL OPERATIONS	PREPARATION FOR INJECTION	INJEC FIRI
TIME (HRS.)	0	.196	.200	2.92	2.92
EVENT	MISSION EVENTS: S-IC PROPULSION SYSTEM IGNITION BEGIN PUSHOVER S-IC NO. 1 ENGINE CUTOFF S-IC NO. 2,3,4 AND 5 ENGINE CUTOFF S-IC SEPARATION S-II ULLAGE ROCKET IGNITION S-II PROPULSION SYSTEM IGNITION C/M LES JETTISON S-II 5 ENGINE CUTOFF S-II SEPARATION S-IVB ULLAGE ROCKETS IGNITION S-IVB PROPULSION SYSTEM IGNITION S-IVB ENGINE CUTOFF EARTH PARKING ORBIT ACHIEVED		MISSION EVENTS: ATTITUDE STABILIZATION POST INJECTION POST INJECTION CHECK SYSTEM CHECKOUT EARTH PARKING ORBIT PARAMETERS CHECK TRANSLUNAR INJECTION PREPARATION TRANSLUNAR INJECTION COUNTDOWN S-IV B PROPULSION SIGNAL		MISSION EVEN S-IVB ULLAG IGNI S-IVB PROPU SYSTE IGNI S-IVB ENGI



PHASE	PARACHUTE DESCENT				
TIME (HRS.)	169.04			168.90	168.9
POST-FLIGHT	IMPACT	RECOVERY AID DEPLOYMENT	DESCENT	SPACECRAFT STABILIZATION	

MISSION EVENTS:
DROGUE PARACHUTE DEPLOYMENT AND OPERATION
MAIN PARACHUTE DEPLOYMENT AND OPERATION
DESCENT SEQUENCE
TOUCHDOWN SEQUENCE

NORTH



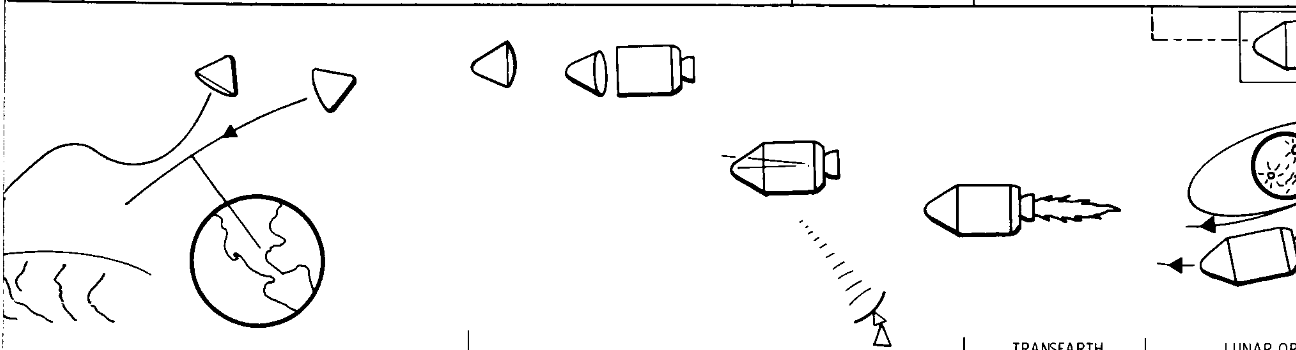
ṲNAR ṮION	TRANSLUNAR COAST				LUNAR ORBIT INJECTION	LUNAR ORBIT PRIOR TO LEM SEPARATION	
ṮION ṮG	C/M LEM TRANSPOSITION	MIDCOURSE CORRECTIONS	TRANSLUNAR OPERATIONS	PREPARATION FOR INJECTION	INJECTION FIRING	ORBITAL OPERATIONS	PREPARATION FOR LEM SEPARATIO
3.01	3.01			67.76	67.76 — 67.85	67.85	

ṮTS:
ṮE ROCKET ṮION
ṮLSION ṮM ṮION
ṮE CUTOFF

MISSION EVENTS:
POST INJECTION CHECK
LEM TRANSPOSITION
PREPARATION FOR COAST ATTITUDE STABILIZATION
TRANSLUNAR COAST PREPARATION
SYSTEMS CHECKOUT
1ST MIDCOURSE CORRECTION
2ND MIDCOURSE CORRECTION
3RD MIDCOURSE CORRECTION
4TH MIDCOURSE CORRECTION
PREPARATION FOR LUNAR ORBIT INJECTION
LUNAR ORBIT INJECTION SIGNAL

MISSION EVENTS:
S/M ULLAGE CONTROL
S/M SPS PROPULSION IGNITION
S/M PROPULSION CUTOFF PERICONE

MISSION EVENTS:
S/M SPS SYSTEM CUTOFF
STABILIZATION
POST INJECTION CHECK
SYSTEMS CHECKOUT
LUNAR ORBIT PARAMETERS CHECK
SCIENTIFIC OBSERVATIONS
PREPARATION FOR LEM SEPARATION



ENTRY			TRANSEARTH COAST			TRANSEARTH INJECTION	LUNAR OR TO LEM
		168.65	168.65		86.24	86.24 — 86.20	86.20
GLIDE	TRANSITION	ENTER	ENTRY ORGANIZATION	TRANSEARTH OPERATIONS	MID COURSE CORRECTIONS	INJECTION FIRING	PREPARATION FOR INJECTION

MISSION EVENTS:
ATTITUDE STABILIZATION BEGIN ENTRY
CONTROL MANEUVERS
TRANSITION TO GLIDE
GLIDE TO PARACHUTE DESCENT
FORWARD HEAT SHIELD JETTISON

MISSION EVENTS:
SPS CUTOFF
POST INJECTION CHECK
SYSTEMS CHECKOUT
1ST MIDCOURSE CORRECTION
2ND MIDCOURSE CORRECTION
3RD MIDCOURSE CORRECTION
4TH MIDCOURSE CORRECTION
PREPARATION FOR ATMOSPHERE ENTRY
SERVICE MODULE JETTISON
BEGIN ATMOSPHERE ENTRY

MISSION EVENTS:
S/M RCS ULLAGE ACCELERATION
SERVICE PROPULSION SYSTEM IGNITION
SYSTEM CUTOFF SERVICE PROPULSION

MISSION EVENTS:
DOCKING
POST DOCKING CHE
SPACECRAFT (C/M + LEM SEPARATI
SYSTEM CHECKOUT
TRANSEARTH INJECT
COUNTDOWN FOR TRANSEARTH I

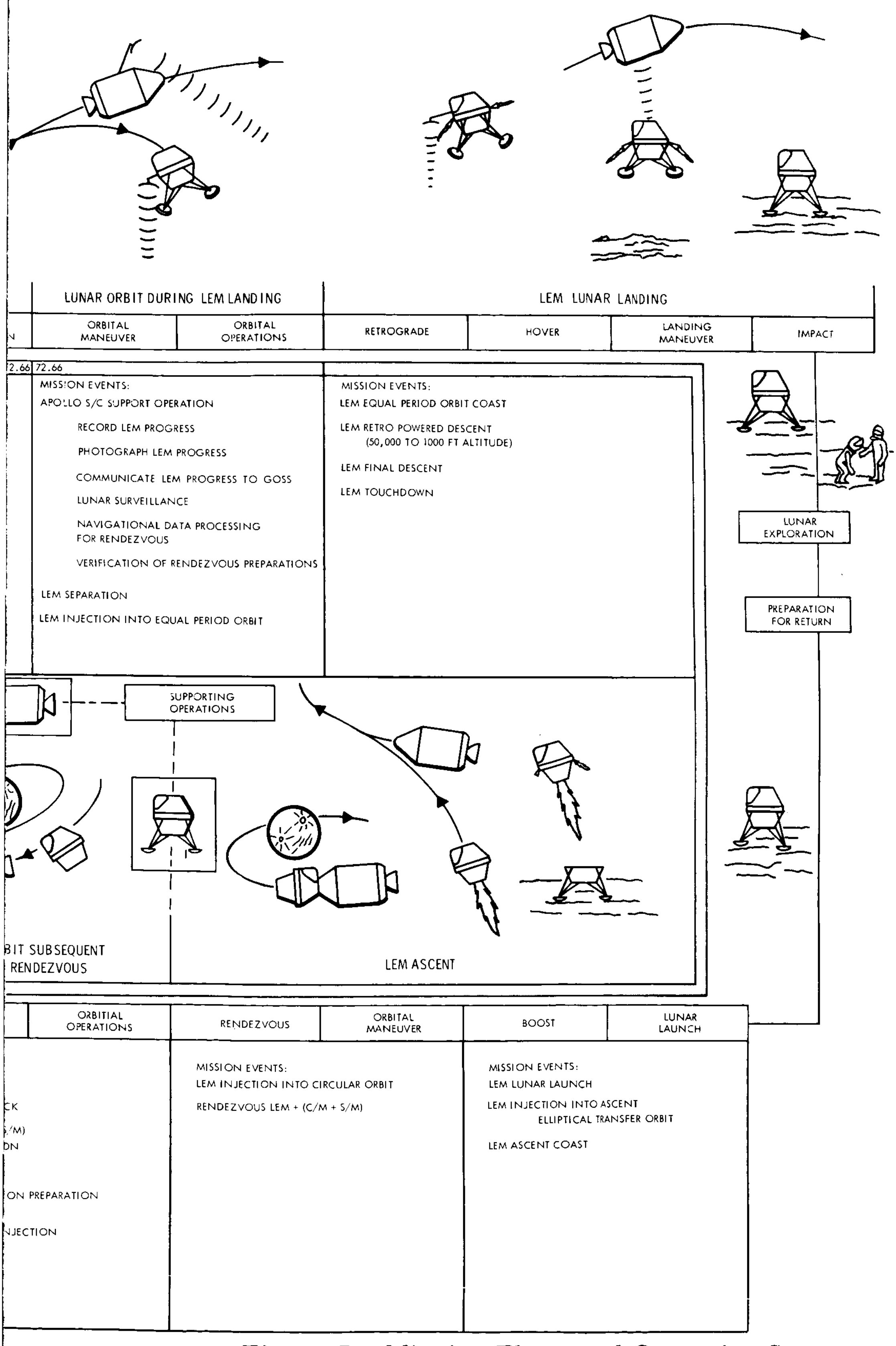


Figure 7. Mission Phase and Operation Segments

SECTION III
MISSION PHASE ANALYSIS

Mission activity has been analyzed by phase for twelve distinct phases of varying time duration. Introductory to each phase analysis, a block diagram indicates the sequence of major events, a diagram presents the geometry involved, and an earth trace for the particular phase is emphasized on the earth trace for the entire mission.

The final figure for each phase is a preliminary time-line analysis of spacecraft system operation. To facilitate analysis, each major spacecraft system is subdivided into quantities called pertinent functions. A pertinent function is an arbitrary grouping of certain hardware and crew procedures that work together to perform a specific system function. The sum total of all of the pertinent functions of a given spacecraft system represents the total operating capability of that system.

Each pertinent function is given a name that is generally understood by technical personnel and is restricted in content for ease of handling. It is expected that it will be possible to develop values for each pertinent function to indicate (1) net reliability of the hardware elements, (2) probability of the functional success, which combines hardware reliability with probabilities of successful crew performance and (3) measures of crew safety at various intervals in the typical mission. Pertinent functions for all spacecraft systems are defined in Appendix F.

The first page of the figure delineates the time occurrence and duration of mission events and requirements during that particular phase. Estimated trajectory data (where applicable) such as altitude, velocity, and spacecraft orientation are also plotted against the time scale. Bar charting is used to indicate the GOSS station which will be in contact with the spacecraft during the mission. The GOSS coverage for the entire mission is summarized in tabular form in Appendix G. The final information on the first page of the figure is a series of bars which indicate when pertinent functions of the Communications and Instrumentation System are performed, and when there is line-of-sight between the GOSS and the Communications and Instrumentation System. It is noted that for various mission phases, only certain of the pertinent functions are involved. Also some of the pertinent functions are seen to be continuous, while others occur in a discrete manner. The second page of the figure for each mission phase is a continuation of the time-line delineation of pertinent functions for various spacecraft systems.

The C/M Structural and Heat Protection System and the S/M Structural System are not included in the spacecraft systems time-line, because their pertinent functions are virtually continuous throughout the entire mission. Similarly, the Controls and Displays System does not appear on the spacecraft systems time-line since the various controls and displays are integral with other spacecraft systems. However, a description of the controls and displays is included in Appendix F.

It is important to point out that the spacecraft system time-line delineation of various pertinent functions is illustrative of how they could be performed during a lunar landing mission; i.e., it is not the only choice that could be made in many cases. Subsequent mission analyses will examine the many trade-offs between systems activity and the available time during each mission phase. Furthermore, in this analysis of a typical lunar landing mission, consideration of spacecraft systems does not include any of their pertinent functions which relate to contingency and/or abort operations. This too, will also be considered in subsequent analyses.

The following explanation refers to the different types of bars which are used in the spacecraft systems time-line analysis.

A solid bar with slanting edges indicates a continuous event. The slanting edges denote that the event both begins and terminates within that time interval.

A solid bar with vertical edges indicates a continuous event which begins and ends abruptly.

A bar composed of individual vertical lines indicates intermittent, discrete events which occur in a short time interval.

A bar composed of individual slanting lines indicates intermittent, discrete events which occur within the time interval limits shown by the slanting lines.

Singular slanting or vertical lines indicate separate short duration events. These lines denote that the event occurs sometime within the tolerance indicated by the slant or width of the line.

Figure 8 explains the reference system for data presented in each phase on the attitude of the spacecraft as it is configured for that particular phase.

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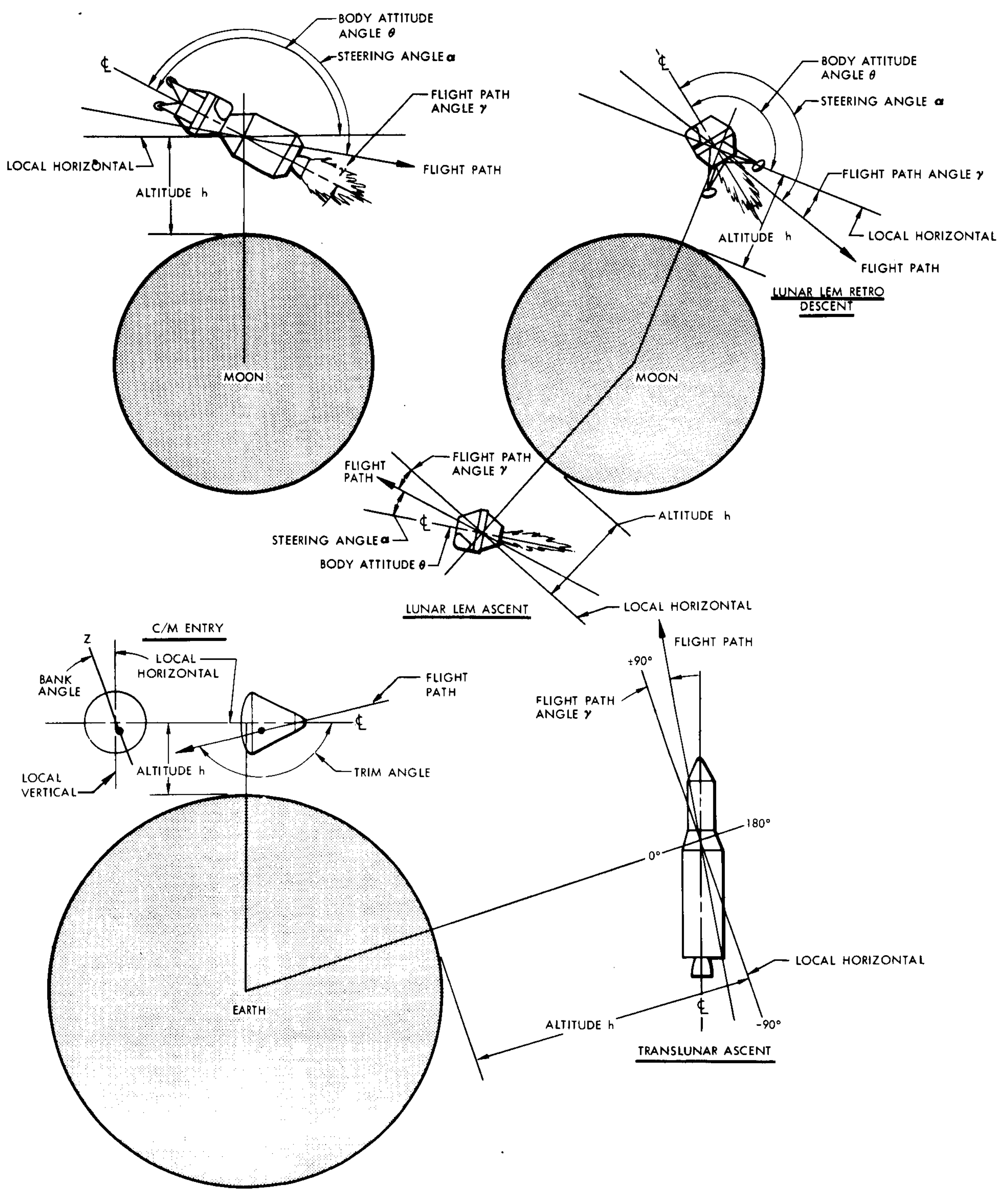


Figure 8. Space Vehicle Flight Attitude Coordinates

~~CONFIDENTIAL~~

[redacted]

ASCENT PHASE

The Ascent Phase begins with S-IC propulsion system ignition (space vehicle liftoff) and ends with S-IVB engine cutoff as the spacecraft and S-IVB are injected into a 100 n.mi. earth parking orbit.

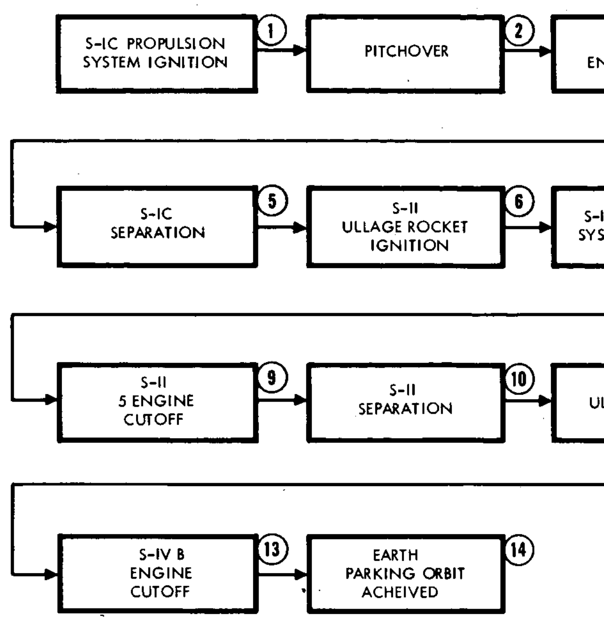
Figure 9 describes the geometry of the Ascent Phase.

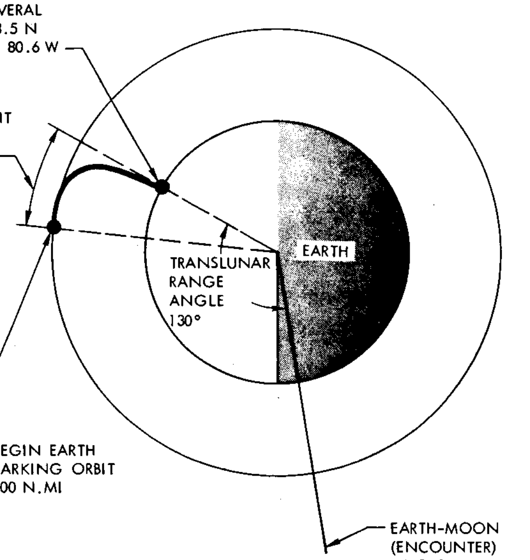
Figure 10 is an earth trace of the Ascent Phase superimposed on a trace for the entire mission.

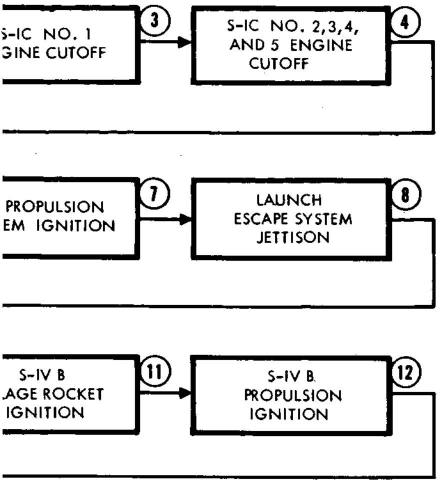
Figure 11 is a two-page time-line delineation of spacecraft system activity during the Ascent Phase.

[redacted]

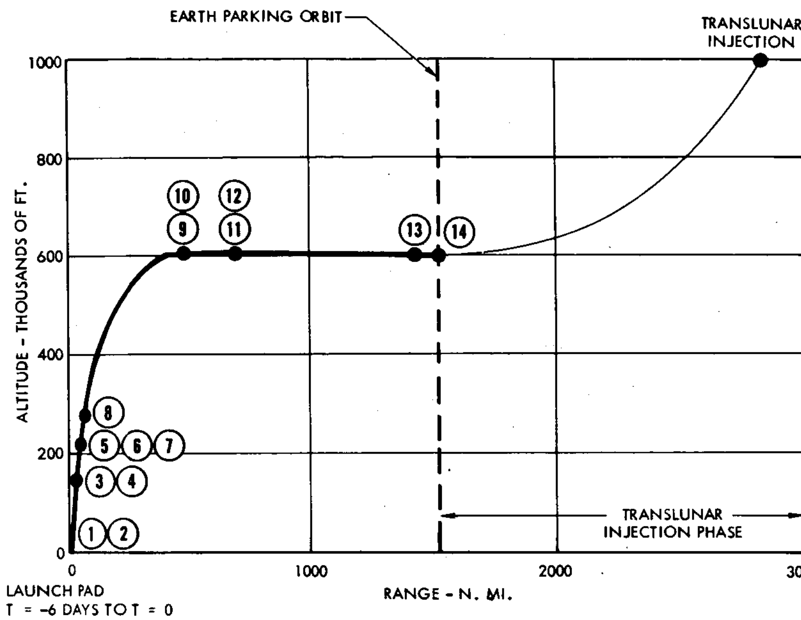
MISSION EVENTS







C-5 CONFIGURATION -
THREE STAGE TO 100.0 N MI EARTH ORBIT
WEIGHT PAYLOAD = 91,525 LBS (SPACECRAFT)



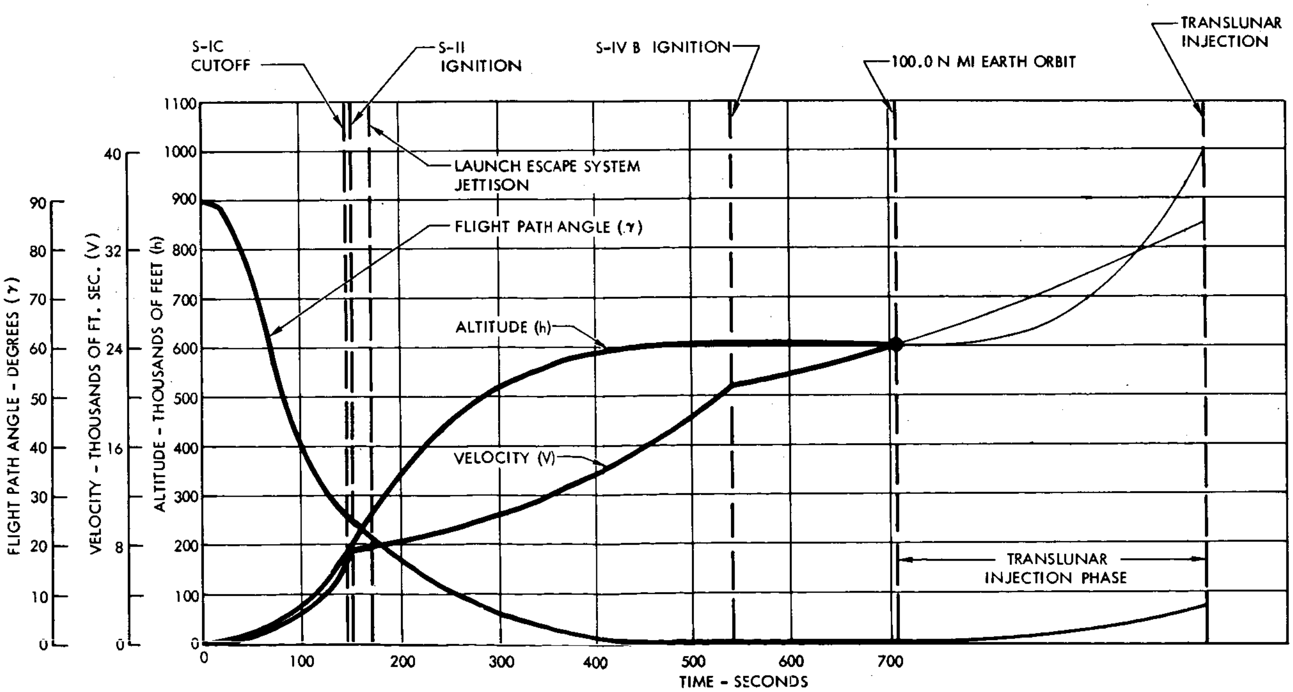


Figure 9. Ascent Phase

[redacted]

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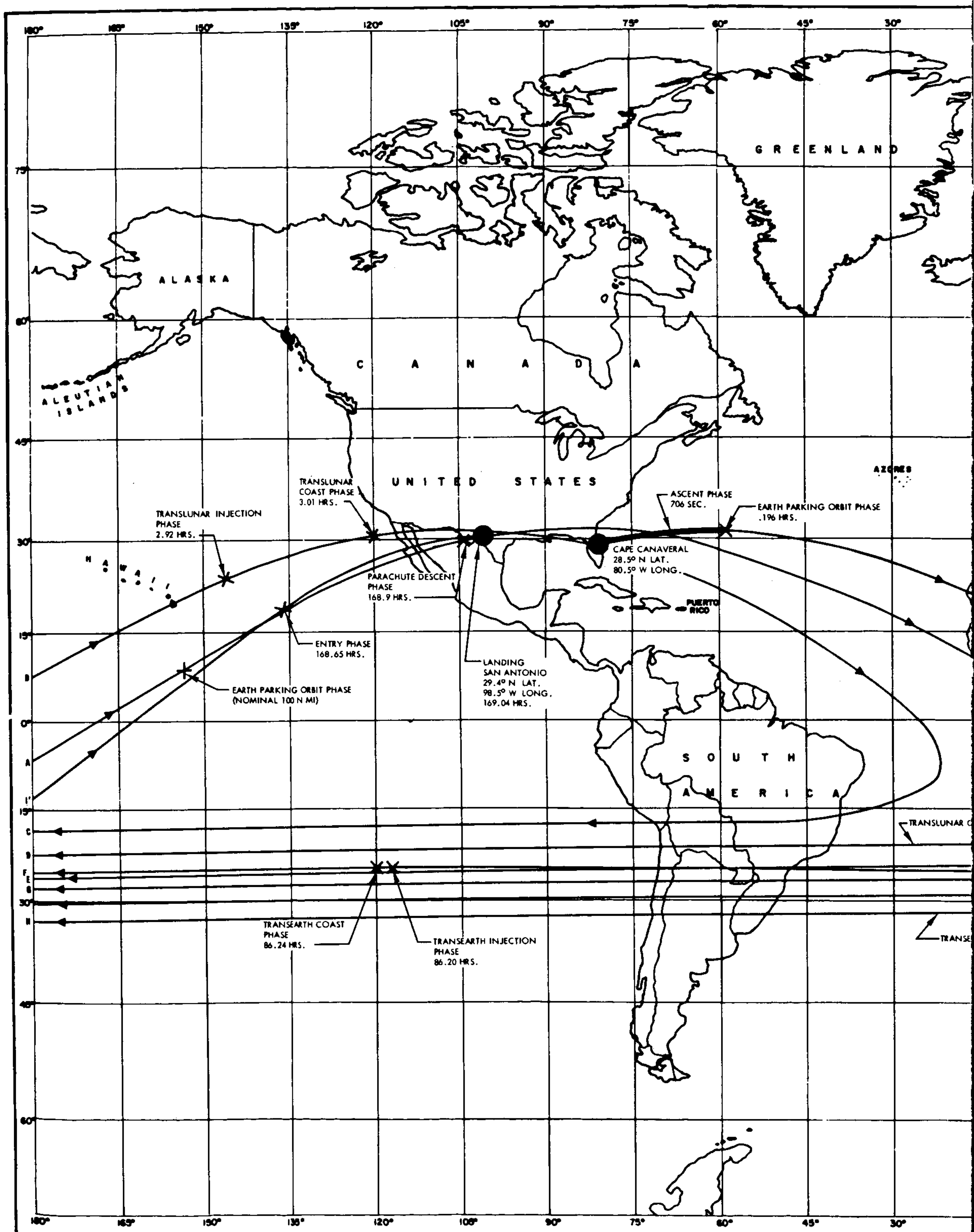


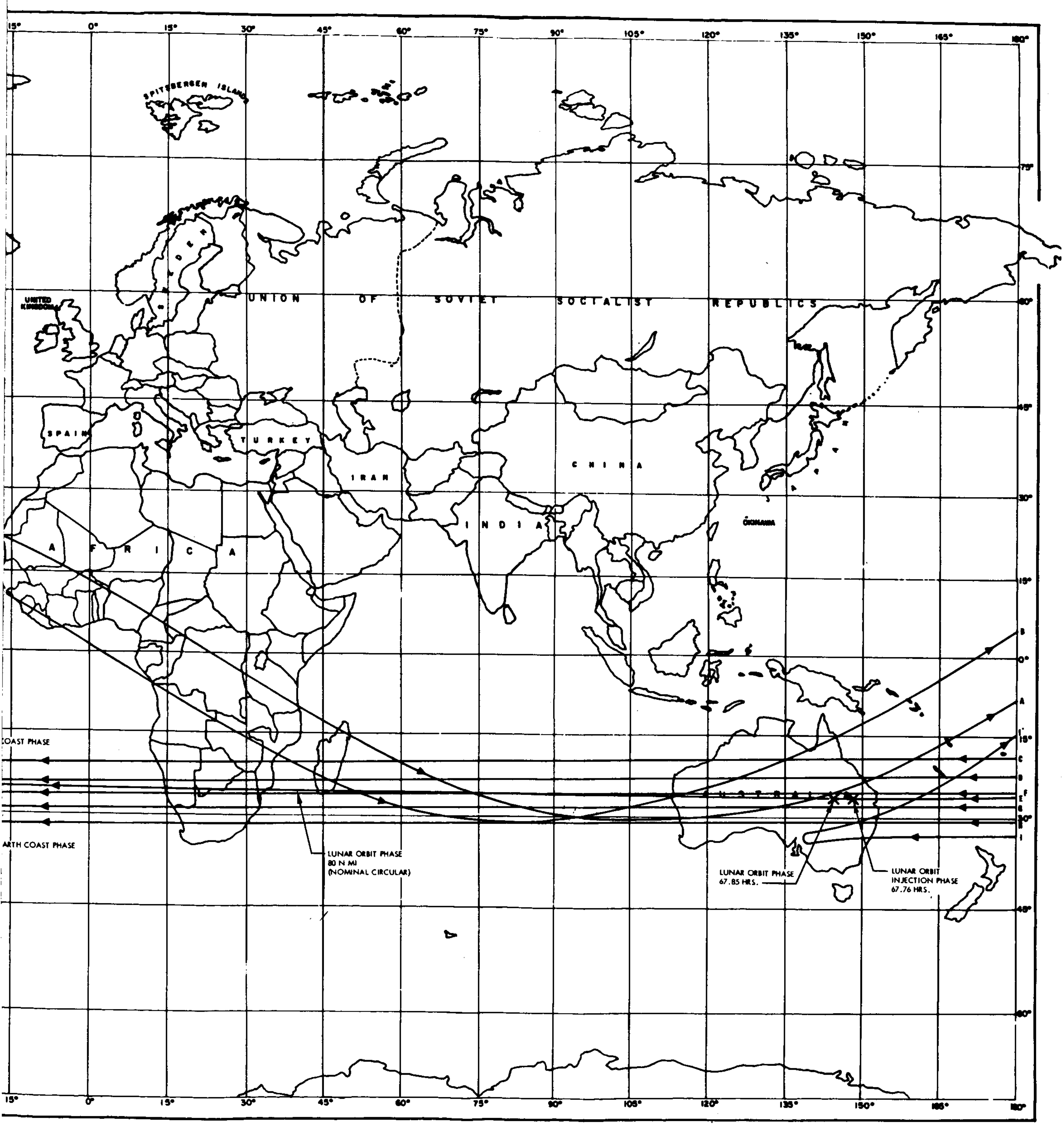


Figure 10. Mission Trajectory Earth Trace-Ascent

[redacted]

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION
S-IC PROPULSION SYSTEM IGNITION & OPERATION	T=.00	143 SECS
BEGIN PITCHOVER & CONTINUE STEERING PROGRAM	T=.003	694 SECS
S-IC NO. 1 ENGINE CUTOFF	T=.040	MILLI SECS
S-IC NO. 2, 3, 4, & 5 ENGINE CUTOFF	T=.041	MILLI SECS
S-II ULLAGE ROCKET IGNITION & OPERATION	T=.041	4 SECS
S-IC SEPARATION SEQUENCE	T=.041	MILLI SECS
S-IC RETRO ROCKET IGNITION	T=.041	2 SECS
S-II PROPULSION SYSTEM IGNITION & OPERATION	T=.041	394 SECS
C/M LES TOWER JETTISON	T=.043	MILLISECS
S-II 5 ENGINE CUTOFF	T=.151	MILLISECS
S-II SEPARATION & RETRO-ROCKET IGNITION	T=.151	2 SECS
S-IVB ULLAGE ROCKETS IGNITIONS & OPERATIONS	T=.152	3 SECS
S-IVB PROPULSION SYSTEM IGNITION & OPERATION	T=.152	159 SECS
S-IVB ENGINE CUTOFF	T=.196	MILLI SECS

TRAJECTORY DATA - ESTIMATED

FLIGHT PATH ANGLE	BODY ATTITUDE	ALTITUDE N MILES	VELOCITY 10^3 FPS	"G" LOAD
0°	0°			4.5
20°	20°	100	24	4
40°	40°	75	18	3
60°	60°	50	12	2
80°	80°	20	6	1
90°	90°			

GOSS COVERAGE - ESTIMATED

CAPE CANAVERAL

BERMUDA

PERTINENT FUNCTIONS

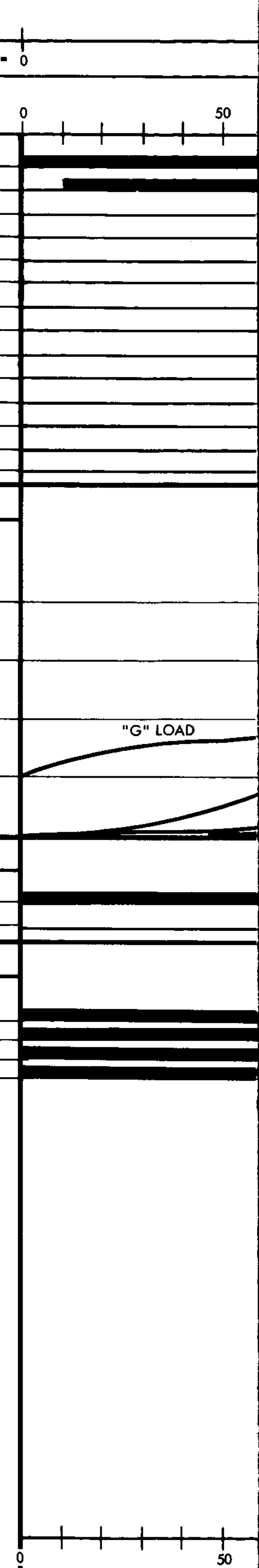
COMMUNICATIONS & INSTRUMENTATION SYSTEM

NEAR EARTH TELEMETRY

NEAR EARTH 2-WAY VOICE WITH GOSS

NEAR EARTH 2-WAY DOPPLER TRACKING/RANGING

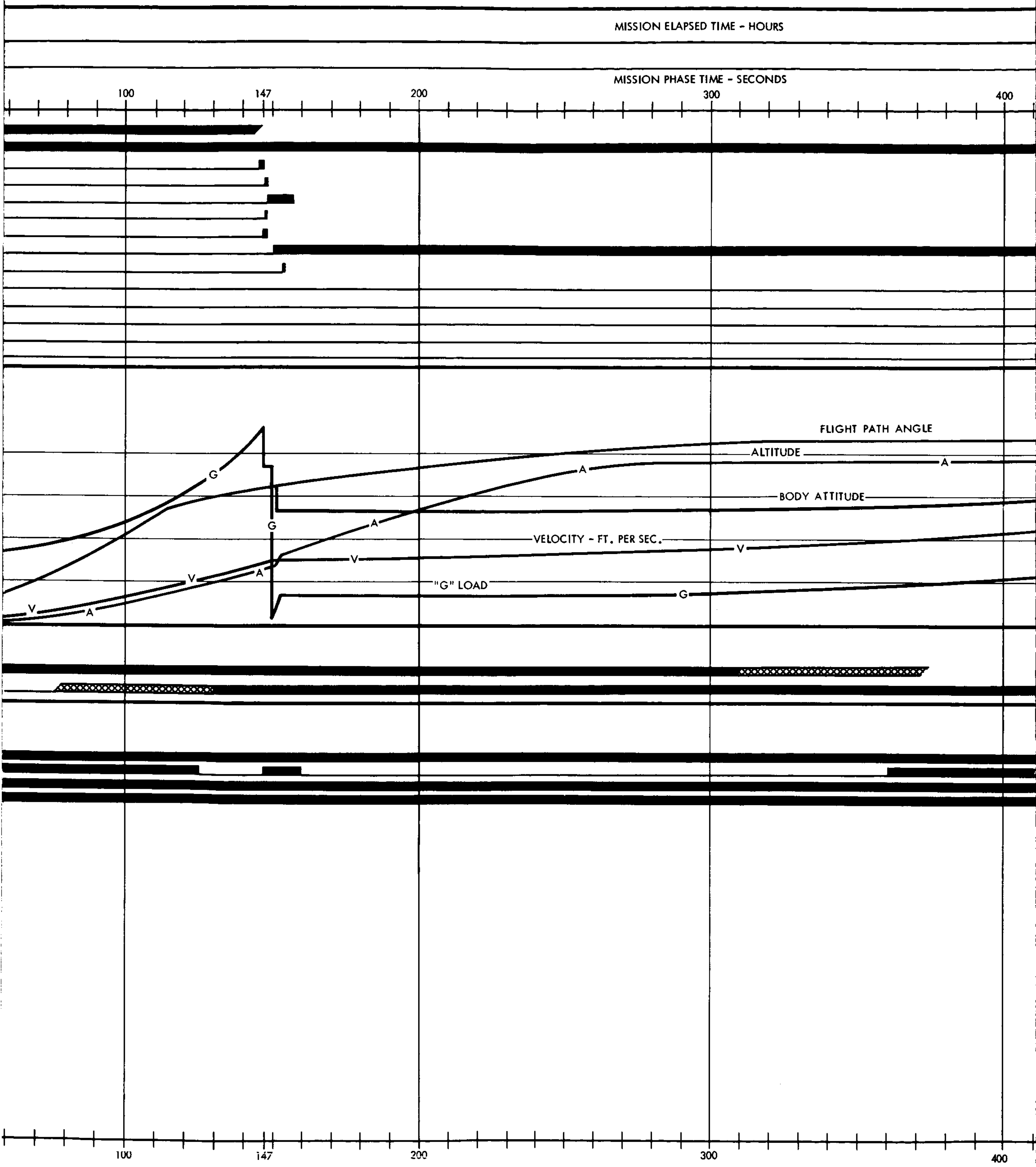
DATA STORAGE RECORDING



GOSS LEGEND

COMMUNICATIONS

RADAR & COMMUNICATIONS



MISSION ELAPSED TIME - HOURS
MISSION PHASE TIME - SECONDS
100
147
200
300
400
FLIGHT PATH ANGLE
ALTITUDE
BODY ATTITUDE
VELOCITY - FT. PER SEC.
"G" LOAD

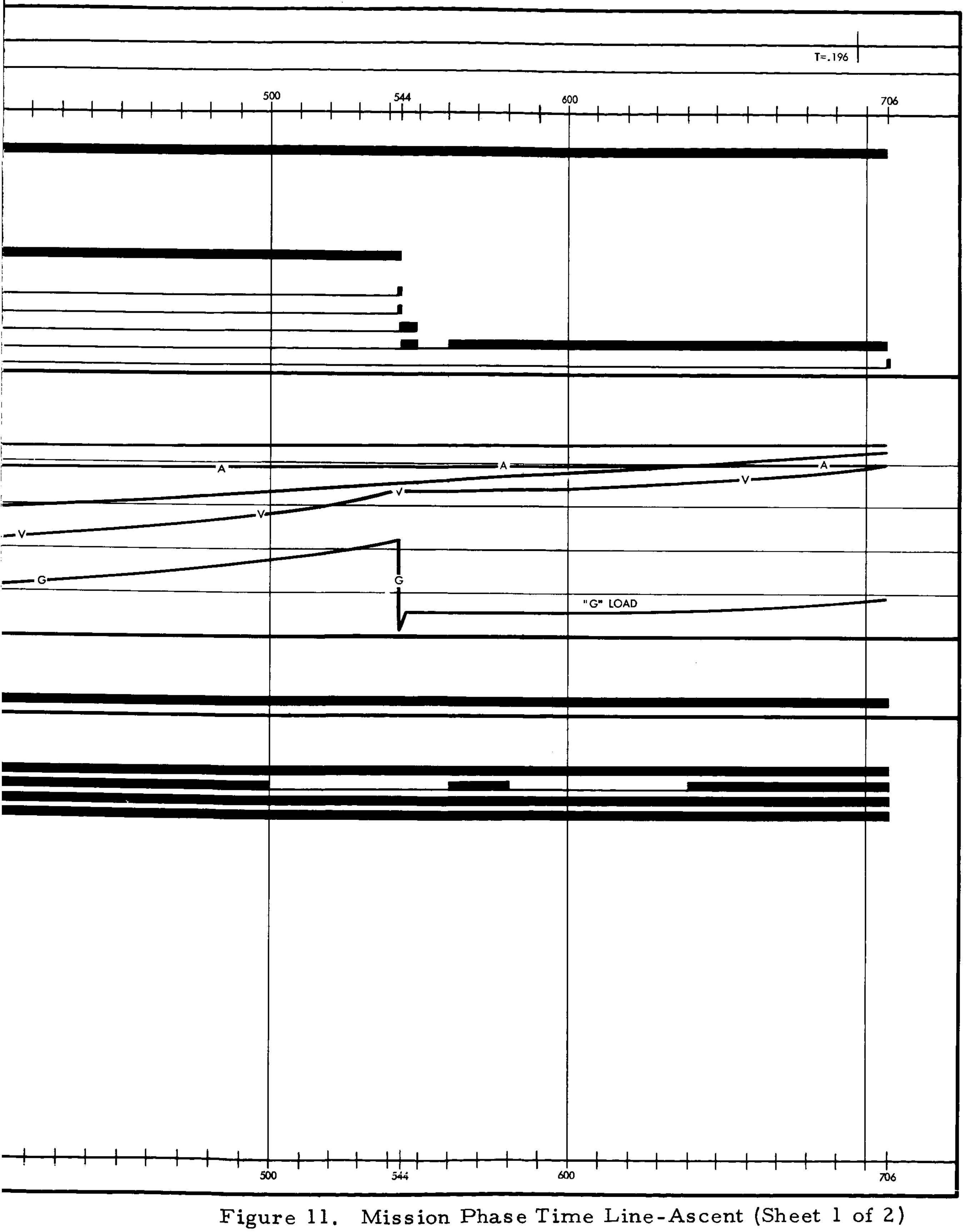


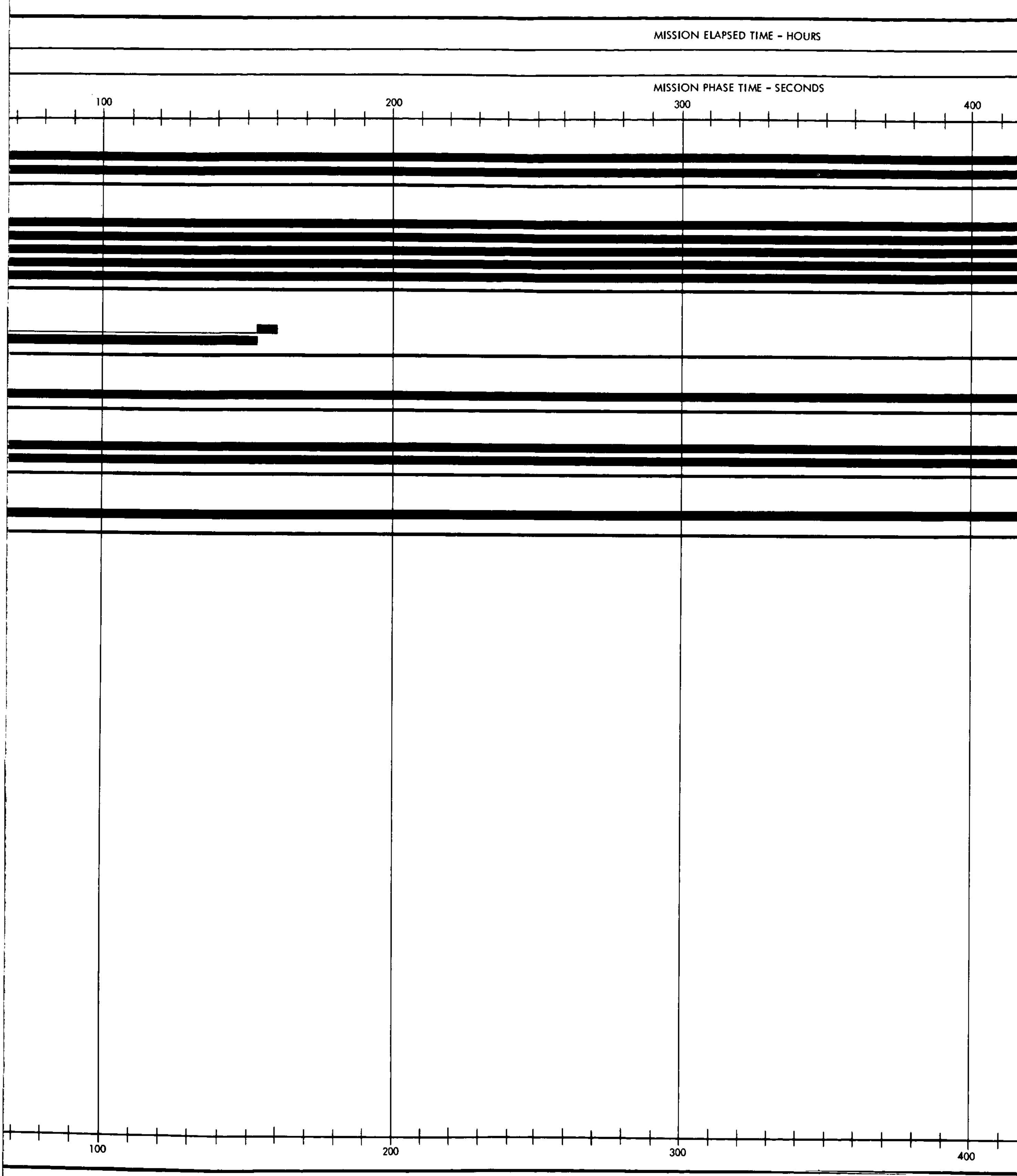
Figure 11. Mission Phase Time Line-Ascent (Sheet 1 of 2)

[redacted]

[redacted]

PERTINENT FUNCTIONS	T = 0
GUIDANCE AND NAVIGATION SYSTEM	
PRIMARY INERTIAL REFERENCE	
SCS MONITOR MODE	
STABILIZATION AND CONTROL SYSTEM	
SECONDARY INERTIAL REFERENCE	
ATTITUDE RATE-OF-CHANGE	
SCS MONITOR MODE	
X AXIS VELOCITY DATA	
TIME DATA	
LAUNCH ESCAPE SYSTEM	
NORMAL JETTISON	
ABORT CAPABILITY	
ENVIRONMENTAL CONTROL SYSTEM	
PRESSURE SUIT ENVIORNMENT	
CREW EQUIPMENT SYSTEM	
CREW SUPPORT & RESTRAINT	
PRESSURE SUIT ENVIRONMENT	
ELECTRICAL POWER SYSTEM	
MAIN POWER - AC & DC	

0 0


MISSION ELAPSED TIME - HOURS
MISSION PHASE TIME - SECONDS
100
200
300
400
100
200
300
400

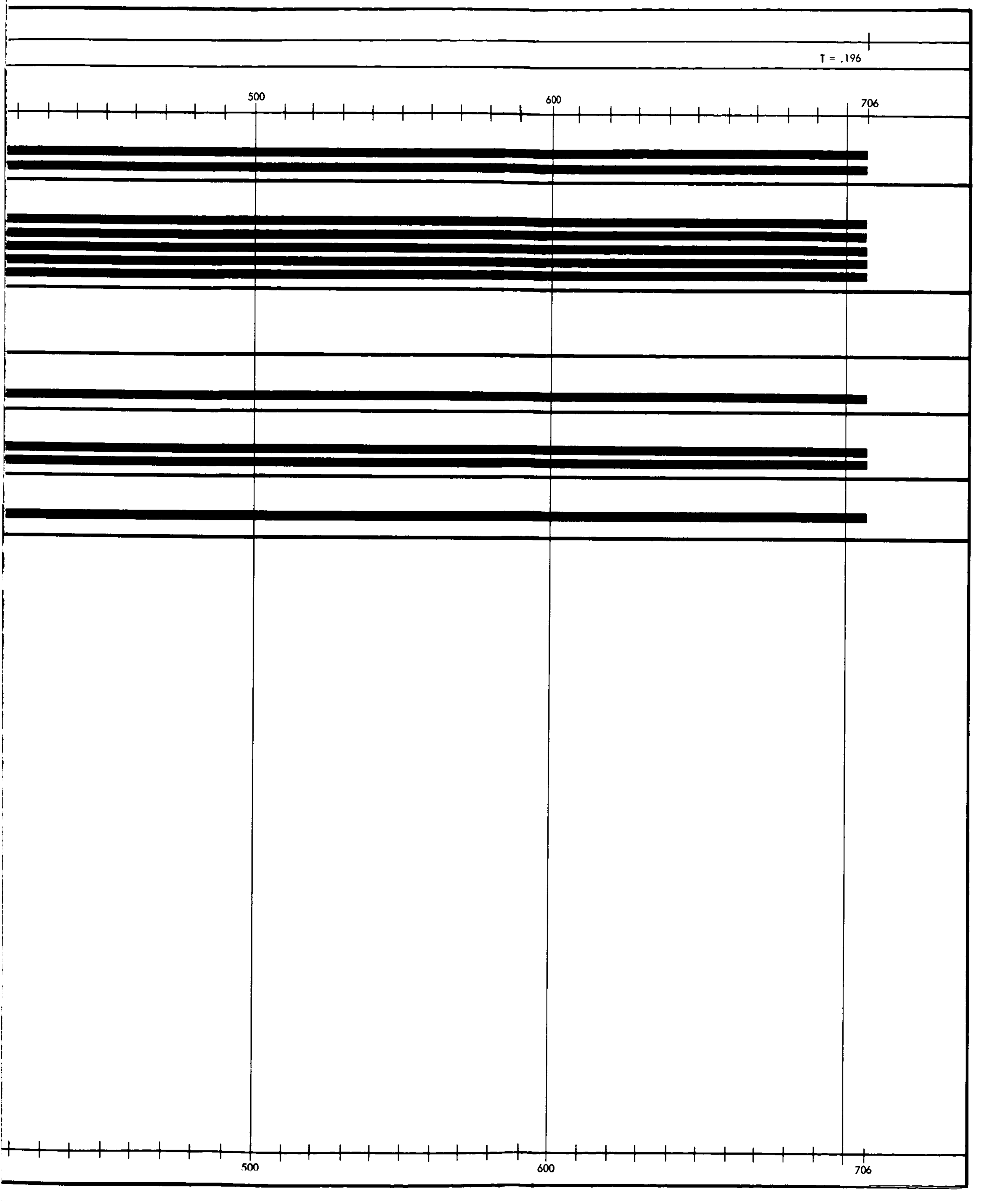


Figure 11. Mission Phase Time Line - Ascent (Sheet 2 of 2)

[redacted]

EARTH PARKING ORBIT PHASE

The Earth Parking Orbit Phase begins with S-IVB engine cutoff as the spacecraft and S-IVB are injected into a 100 n. mile earth parking orbit. The phase ends with S-IVB ullage rocket ignition for translunar injection.

Figure 12 describes the geometry of the Earth Parking Orbit Phase.

Figure 13 is an earth trace of the Earth Parking Orbit Phase superimposed on a trace for the entire mission.

Figure 14 is a two-page time-line delineation of spacecraft system activity during the Earth Parking Orbit Phase.

[redacted]

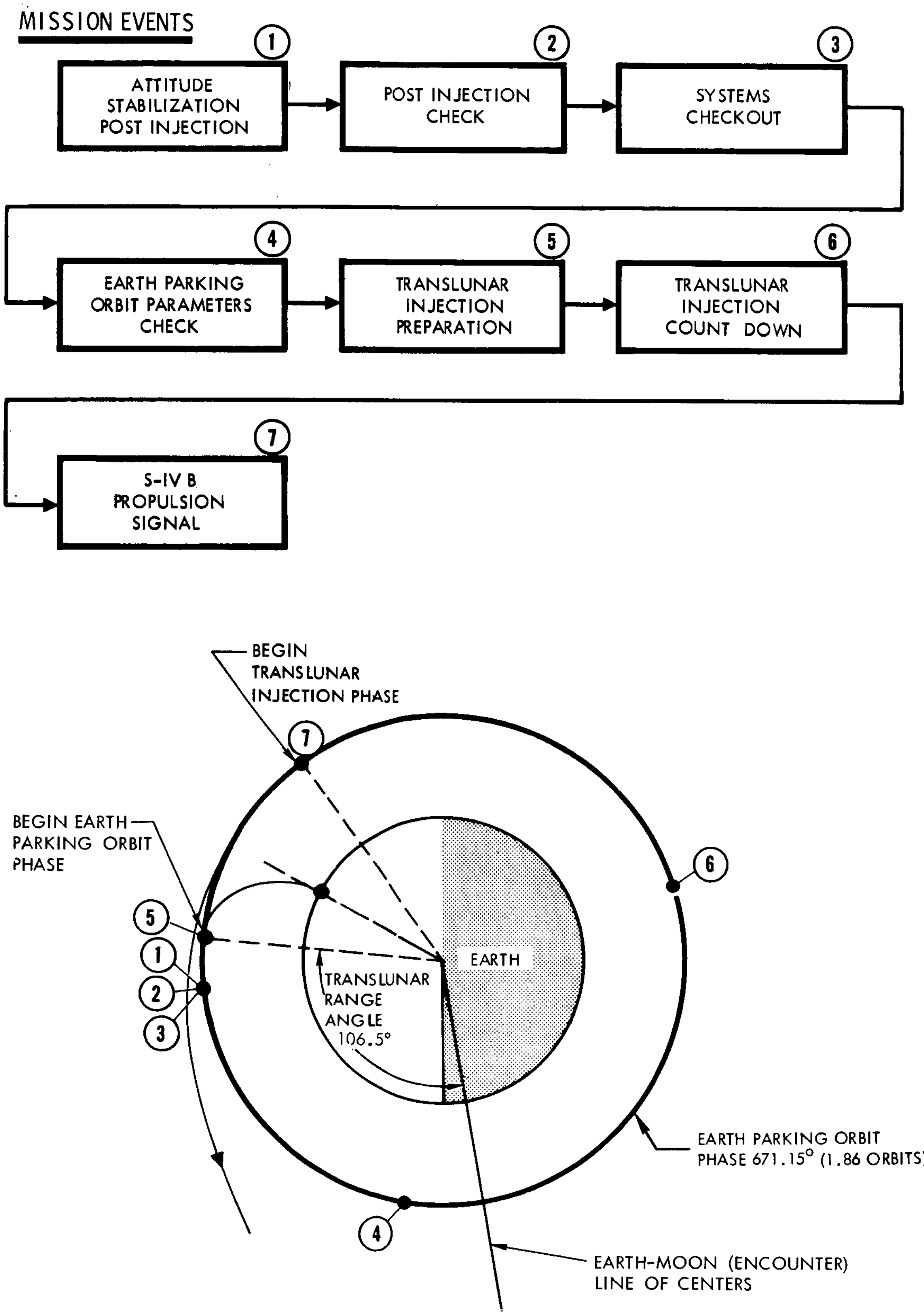
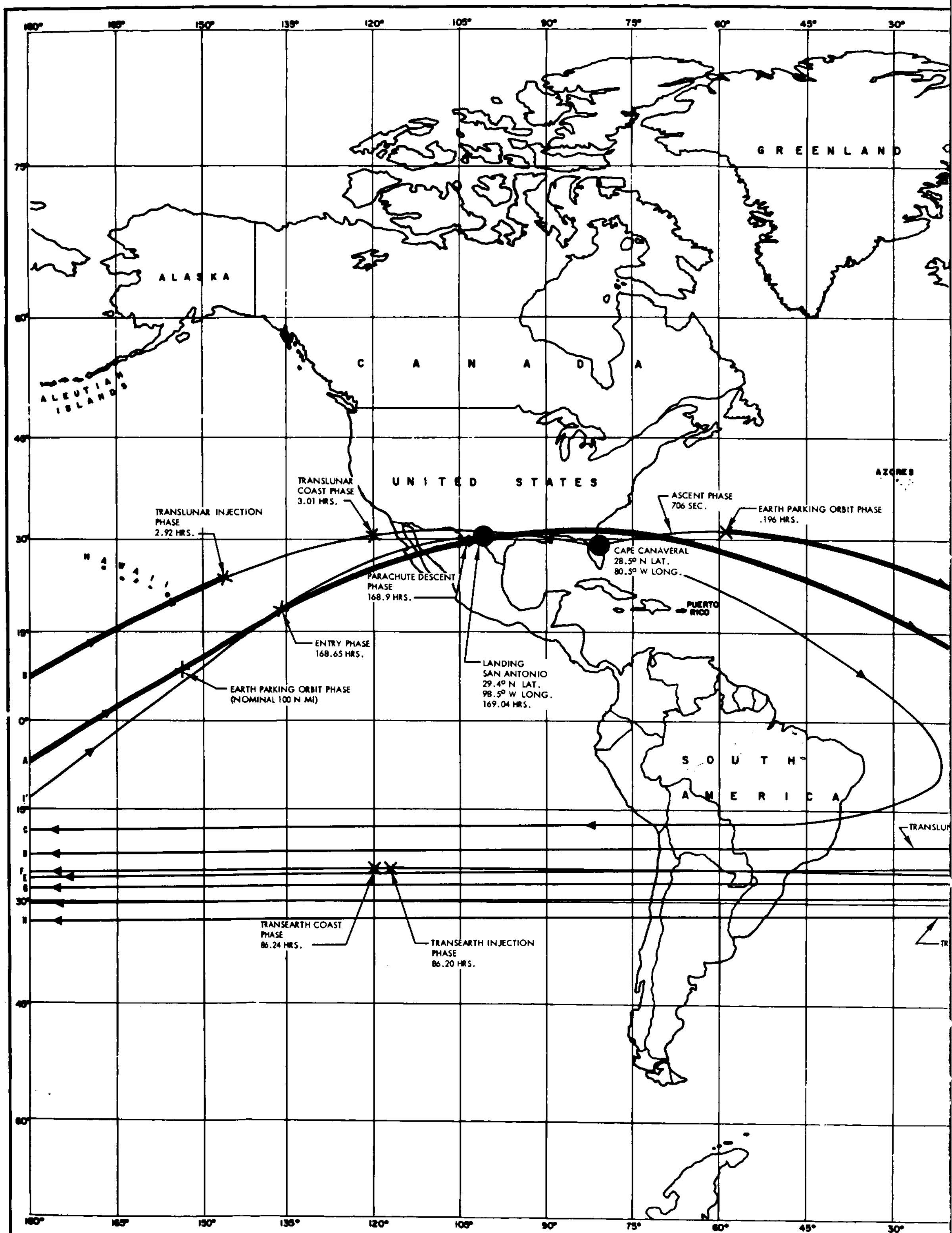


Figure 12. Earth Parking Orbit Phase

[redacted]



GREENLAND
ALASKA
CANADA
ALEUTIAN ISLANDS
UNITED STATES
AZORES
HAWAII
PUERTO RICO
SOUTH AMERICA
TRANSLUNAR COAST PHASE
3.01 HRS.
TRANSLUNAR INJECTION PHASE
2.92 HRS.
ASCENT PHASE
706 SEC.
EARTH PARKING ORBIT PHASE
.196 HRS.
CAPE CANAVERAL
28.5° N LAT.
80.5° W LONG.
PARACHUTE DESCENT PHASE
168.9 HRS.
ENTRY PHASE
168.65 HRS.
LANDING
SAN ANTONIO
29.4° N LAT.
98.5° W LONG.
169.04 HRS.
EARTH PARKING ORBIT PHASE
(NOMINAL 100 N MI)
TRANSEARTH COAST PHASE
86.24 HRS.
TRANSEARTH INJECTION PHASE
86.20 HRS.
180° 165° 150° 135° 120° 105° 90° 75° 60° 45° 30°
75° 60° 45° 30° 15° 0° 15° 30° 45° 60°

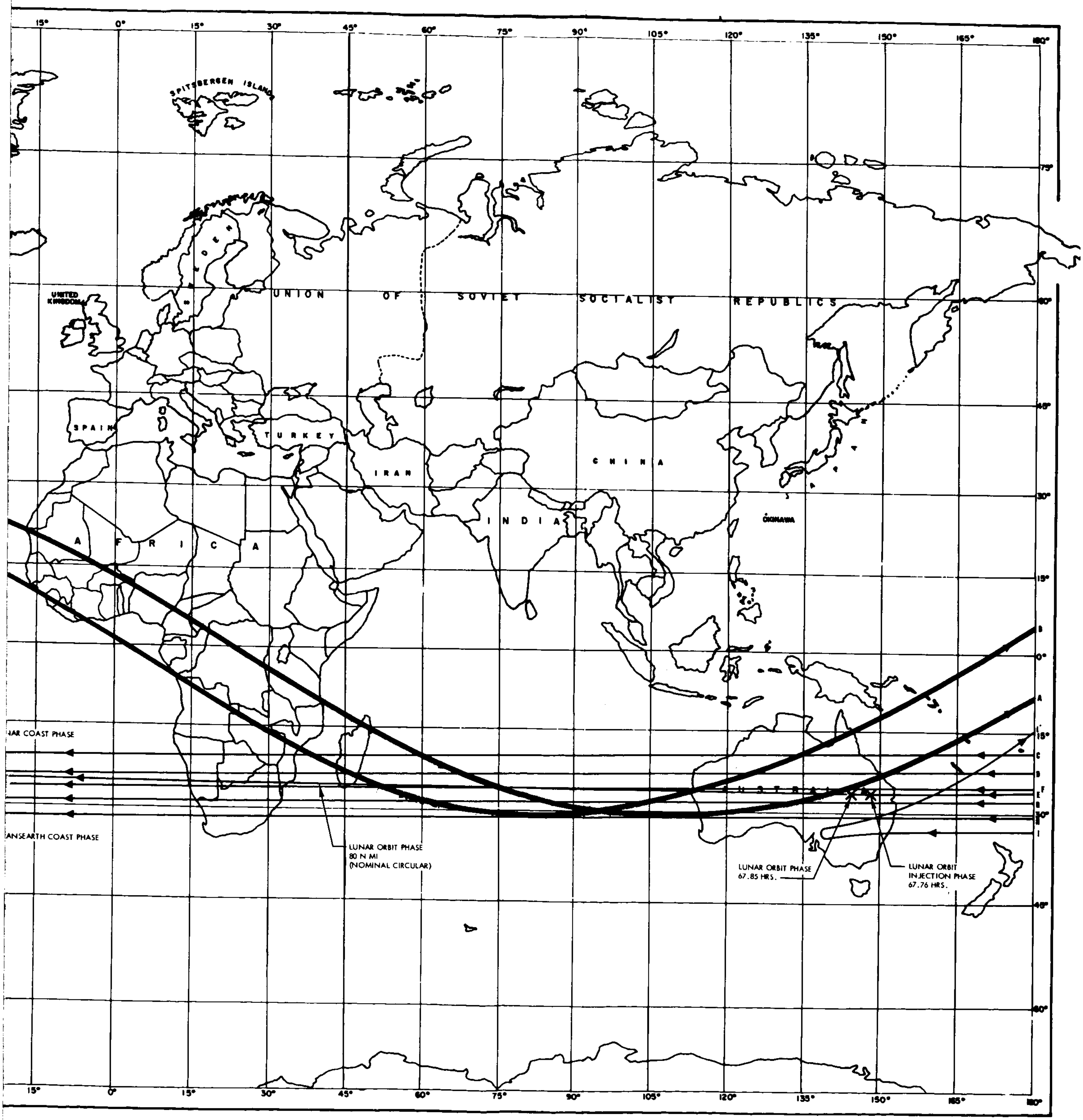
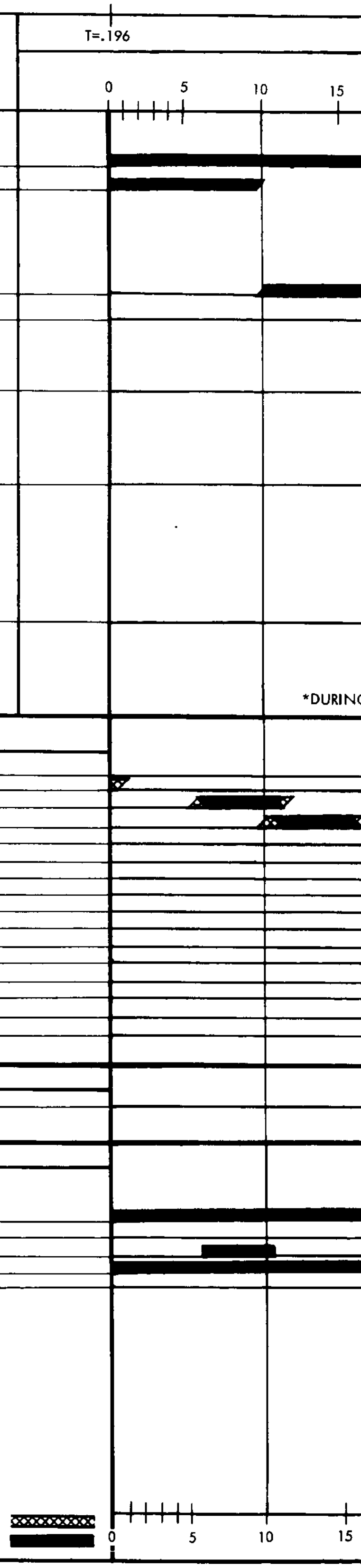


Figure 13. Mission Trajectory Earth Trace-Earth Parking Orbit

[redacted]

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION
S-IVB PROPULSION CUTOFF.	T = .196	
ATTITUDE STABILIZATION (S-IVB G & C).		163 MIN
POST EARTH ORBIT CHECK & CONFIGURATION. CONTROLS SET FOR EARTH ORBIT PHASE. INSTRUMENTATION READOUT CHECK. BIOLOGICAL & RADIATION CHECK. CREW & EQUIPMENT ARRANGEMENT.	T = .196	10 MIN
SYSTEMS CHECKOUT	T = .346	15 MIN
EARTH ORBIT PARAMETERS CHECK. ON BOARD TRAJECTORY DATA PROCESSING. MANUAL TRAJECTORY DATA PROCESSING. GOSS TRAJECTORY DATA PROCESSING.	T = .596	75 MIN
TRANSLUNAR INJECTION PREPARATIONS. INJECTION PARAMETERS DATA PROCESSING. IMU FINE ALIGNMENT.	T = 1.92	30 MIN
TRANSLUNAR INJECTION COUNTDOWN. SYSTEMS SET-UP EQUIPMENT SECURED. CREW SECURED. *GOSS READINESS VERIFIED.	T = 2.42	30 MIN
S-IVB PROPULSION SIGNAL	T = 2.92	



GOSS COVERAGE - ESTIMATED

CAPE CANAVERAL
BERMUDA
GRAND CANARY ISLANDS
KANO
ZANZIBAR
JOHANNESBURG
INDIAN OCEAN SHIP
MUCHEA
WOOMERA
DSIF WOOMERA
CANTON
KAUAI
POINT ARGUELLO
DSIF GOLDSTONE
GUAYMAS
HOUSTON

POSITIONAL DATA - ESTIMATED

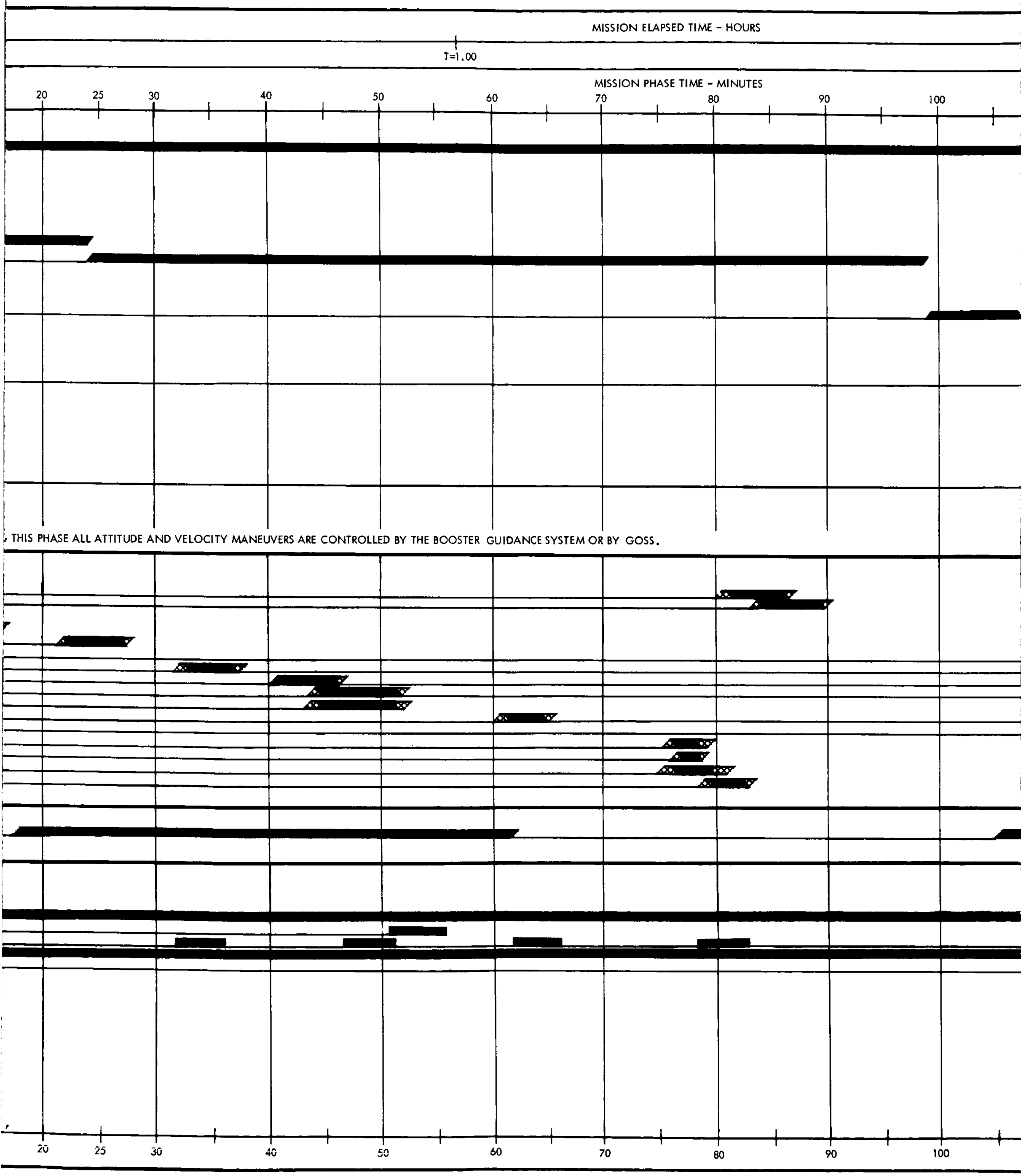
OVER EARTH SHADOW

PERTINENT FUNCTIONS

COMMUNICATIONS & INSTRUMENTATION SYSTEM

NEAR EARTH TELEMETRY
C/M DSIF TV TRANSMISSION
NEAR EARTH 2-WAY VOICE WITH GOSS
NEAR EARTH 2-WAY DOPPLER TRACKING/RANGING
DSIF DOPPLER TRACKING/RANGING

GOSS LEGEND:
COMMUNICATIONS
RADAR & COMMUNICATIONS



MISSION ELAPSED TIME - HOURS
T=1.00
MISSION PHASE TIME - MINUTES
20 25 30 40 50 60 70 80 90 100
THIS PHASE ALL ATTITUDE AND VELOCITY MANEUVERS ARE CONTROLLED BY THE BOOSTER GUIDANCE SYSTEM OR BY GOSS.
20 25 30 40 50 60 70 80 90 100

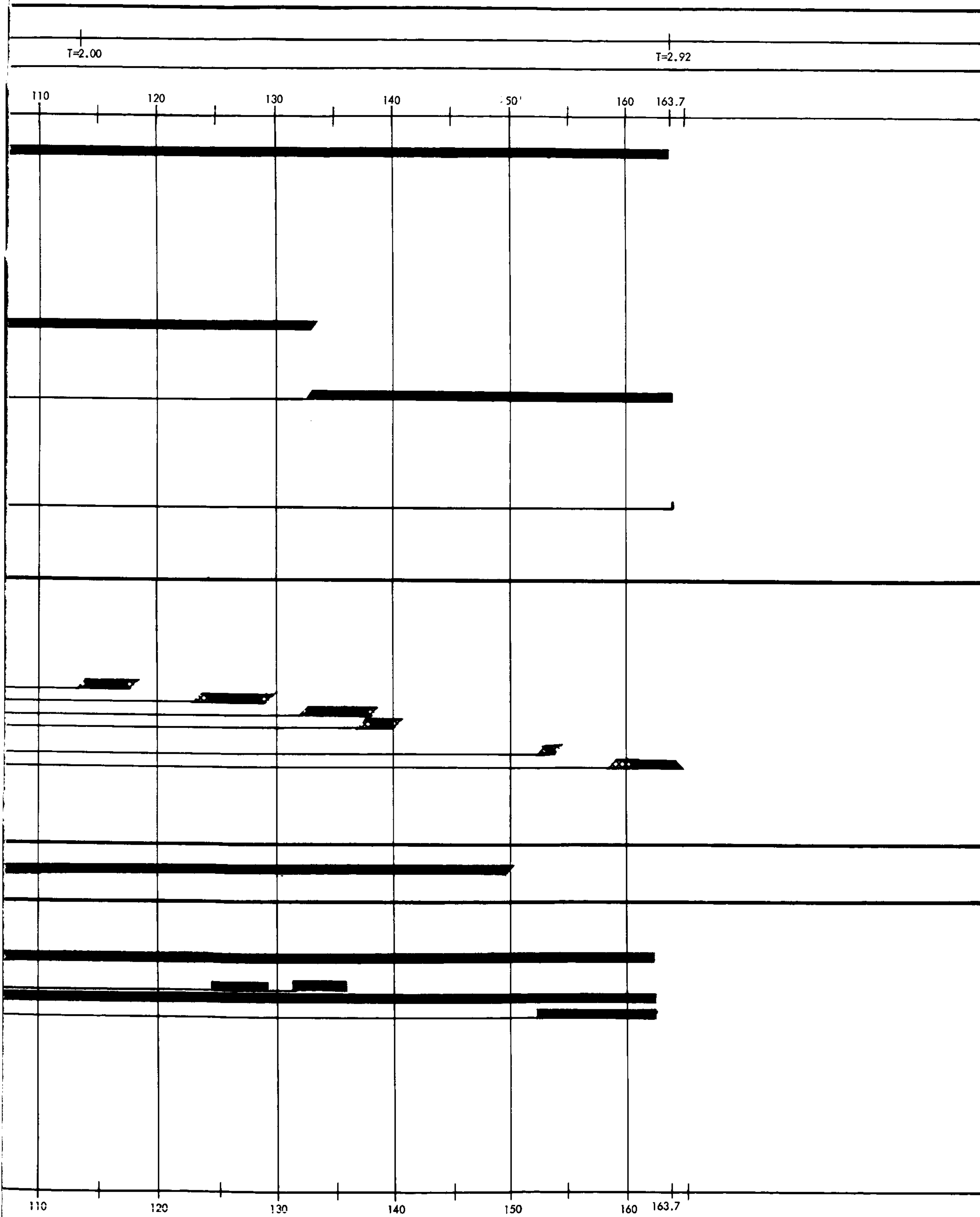


Figure 14. Mission Phase Time Line - Earth Parking Orbit (Sheet 1 of 2)

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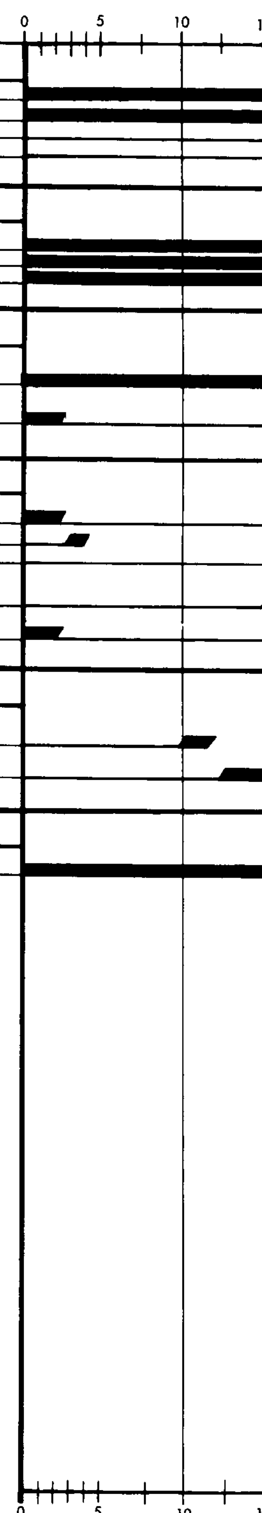
PERTINENT FUNCTIONS

T = .196



GUIDANCE AND NAVIGATION SYSTEM

PRIMARY INERTIAL REFERENCE
SCS MONITOR MODE
EARTH PARKING ORBIT AND EPHEMERIDES
TRANSLUNAR INJECTION PARAMETERS

STABILIZATION AND CONTROL SYSTEM

SECONDARY INERTIAL REFERENCE
ATTITUDE RATE-OF-CHANGE
SCS MONITOR MODE

ENVIRONMENTAL CONTROL SYSTEM

"SHIRTSLEEVE" ENVIRONMENT

PRESSURE SUIT ENVIRONMENT

CREW EQUIPMENT SYSTEM

CREW SUPPORT & RESTRAINT
REPOSITION CENTER COUCH
REPLACE CENTER COUCH

HYGIENE & HEALTH FUNCTION

PRESSURE SUIT ENVIRONMENT

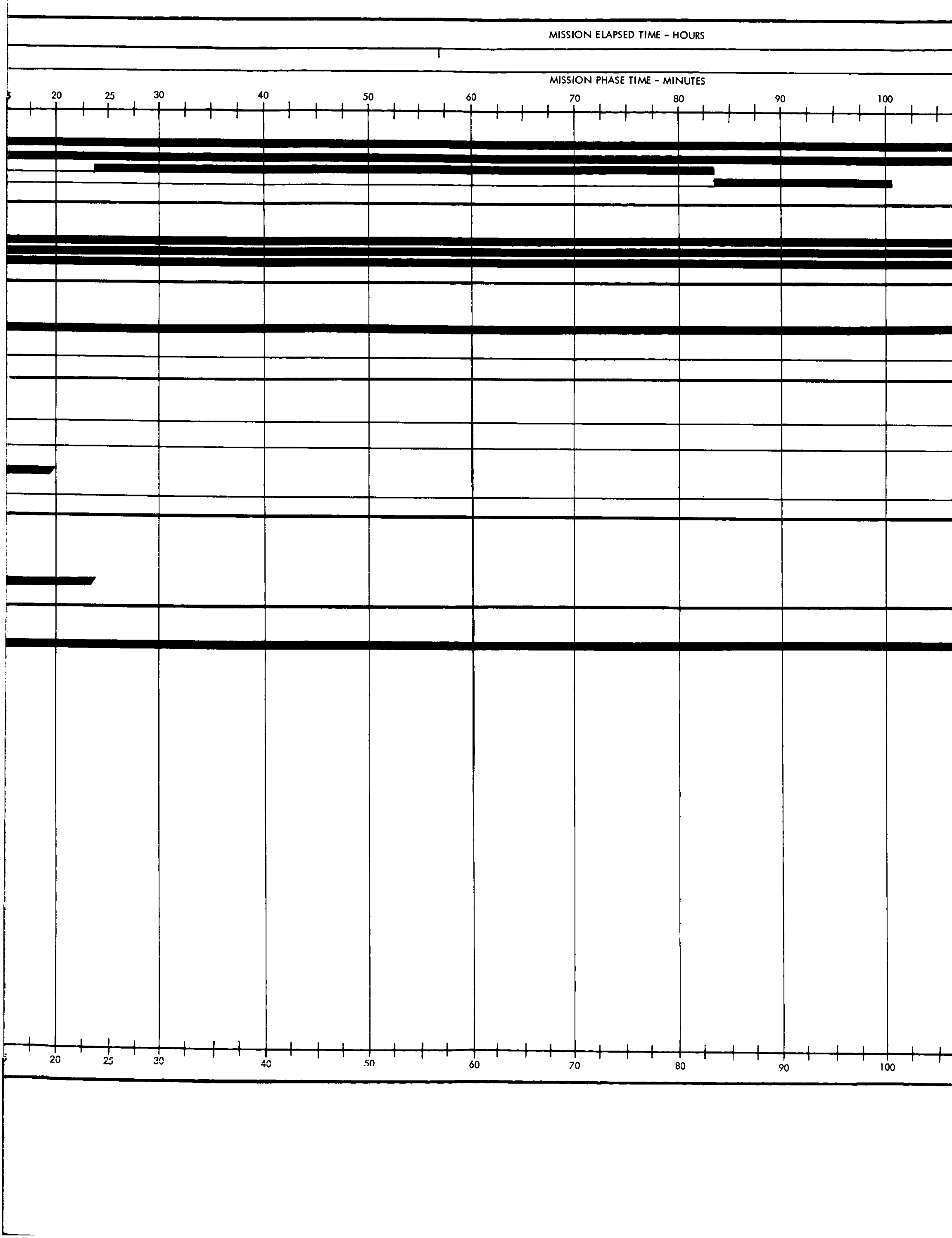
IN-FLIGHT TEST SYSTEM

AUTOMATIC SYSTEMS CHECKOUT

MANUAL SYSTEMS CHECKOUT

ELECTRICAL POWER SYSTEM

MAIN POWER - AC & DC



MISSION ELAPSED TIME - HOURS
MISSION PHASE TIME - MINUTES
20 25 30 40 50 60 70 80 90 100
20 25 30 40 50 60 70 80 90 100

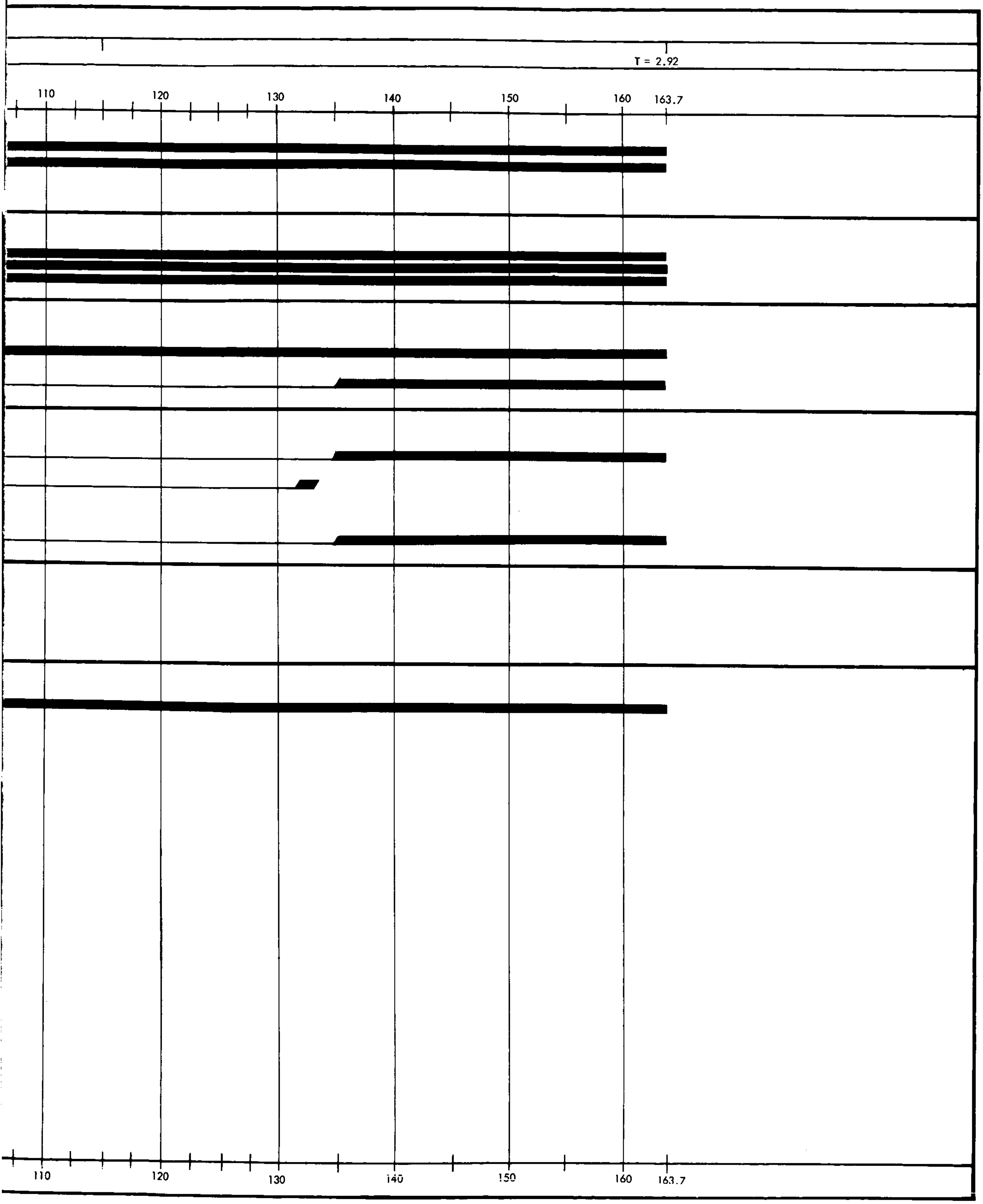


Figure 14. Mission Phase Time Line - Earth Parking Orbit (Sheet 2 of 2)

[redacted]

[redacted]

TRANSLUNAR INJECTION PHASE

The Translunar Injection Phase begins with S-IVB ullage rocket ignition and ends with S-IVB engine cutoff.

Figure 15 describes the geometry of the Translunar Injection Phase.

Figure 16 is an earth trace of the Translunar Injection Phase superimposed on a trace for the entire mission.

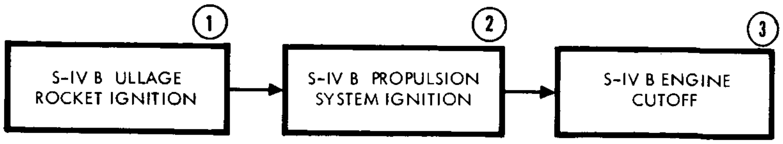
Figure 17 is a two-page time-line delineation of spacecraft system activity during the Translunar Injection Phase.

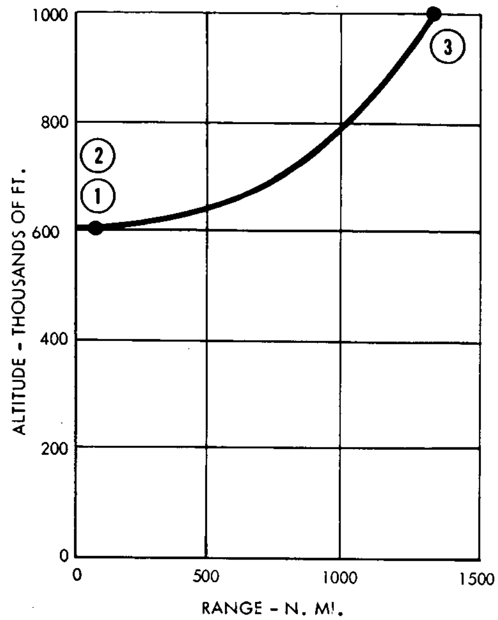
[redacted]

BEGIN
COAST

[redacted]

MISSION EVENTS

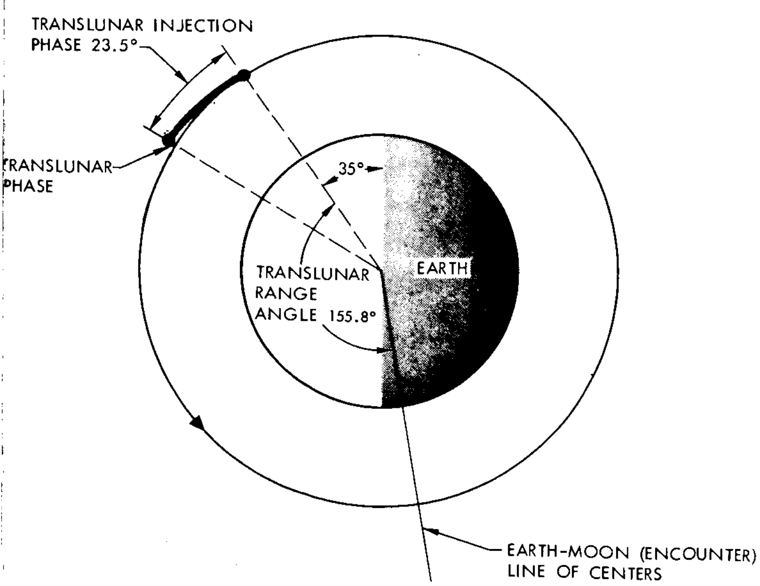




TRANSLUNAR INJECTION VIA 100.0 N MI EARTH ORBIT

S-IV B THIRD STAGE + SPACECRAFT

WEIGHT PAYLOAD = 91,525 LBS (SPACECRAFT)



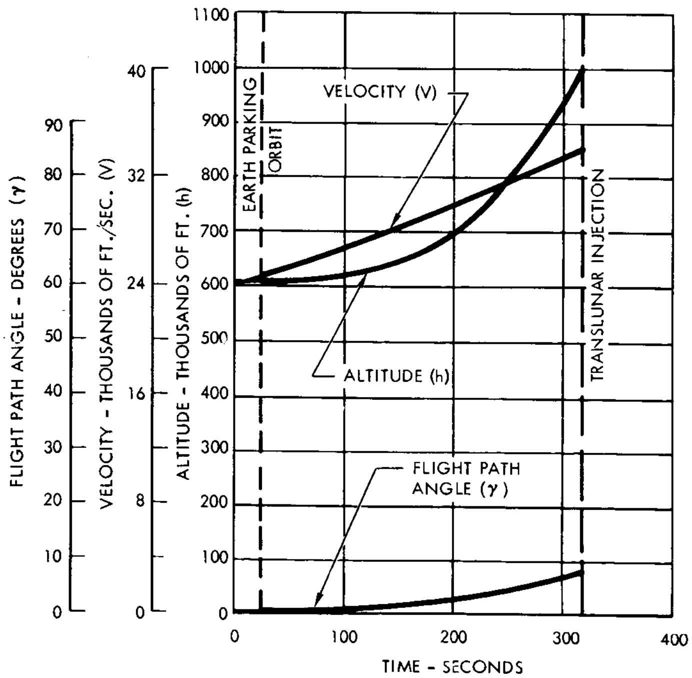
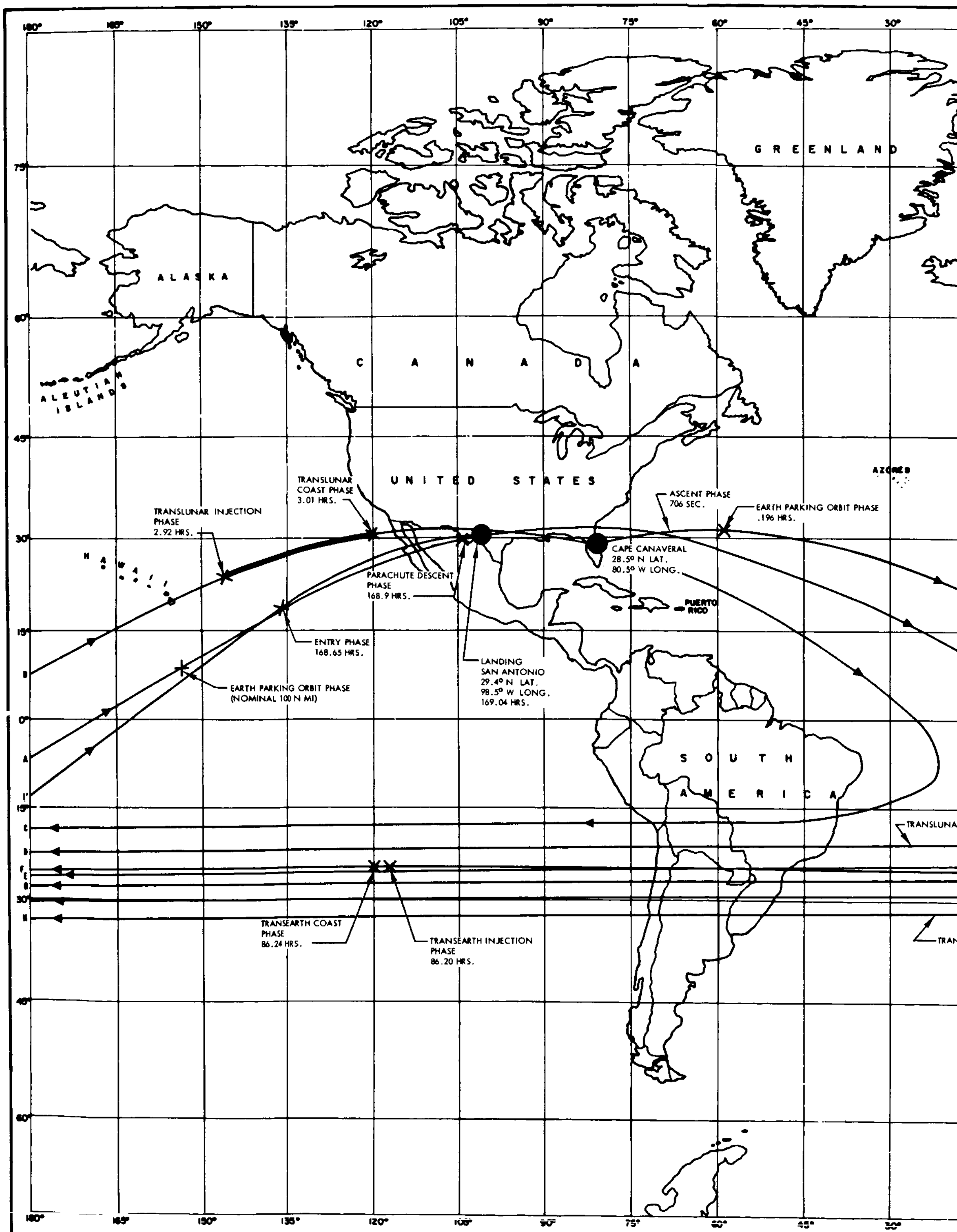


Figure 15. Translunar Injection Phase

[redacted]

[redacted]



180°
165°
150°
135°
120°
105°
90°
75°
60°
45°
30°
75°
60°
45°
30°
15°
0°
15°
30°
45°
60°
GREENLAND
ALASKA
CANADA
ALEUTIAN ISLANDS
UNITED STATES
AZORES
HAWAII
PUERTO RICO
SOUTH AMERICA
TRANSLUNAR COAST PHASE 3.01 HRS.
TRANSLUNAR INJECTION PHASE 2.92 HRS.
ASCENT PHASE 706 SEC.
EARTH PARKING ORBIT PHASE .196 HRS.
CAPE CANAVERAL 28.5° N LAT. 80.5° W LONG.
PARACHUTE DESCENT PHASE 168.9 HRS.
ENTRY PHASE 168.65 HRS.
EARTH PARKING ORBIT PHASE (NOMINAL 100 N MI)
LANDING SAN ANTONIO 29.4° N LAT. 98.5° W LONG. 169.04 HRS.
TRANSEARTH COAST PHASE 86.24 HRS.
TRANSEARTH INJECTION PHASE 86.20 HRS.
B
A
I'
C
D
F
E
G
H

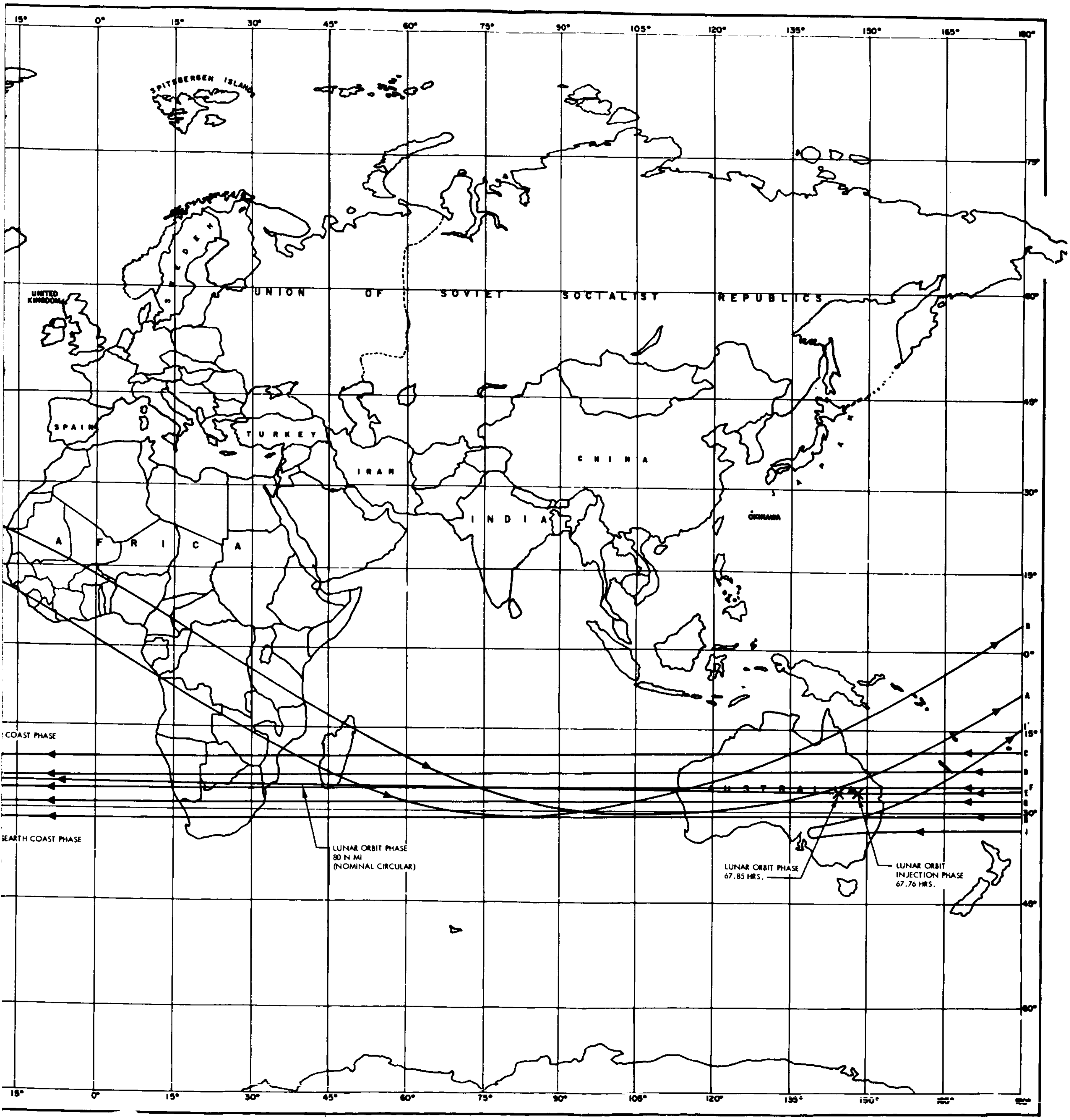


Figure 16. Mission Trajectory Earth Trace-Translunar Injection

 [redacted]

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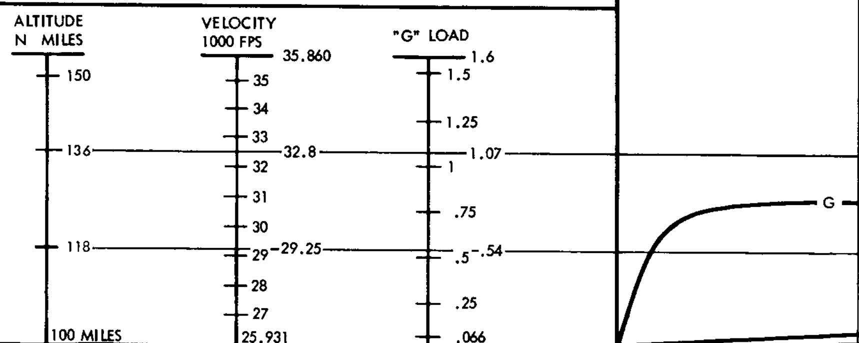
MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION
S-IVB ULLAGE ACCELERATION	T = 2.92	4 SEC
S-IVB PROPULSION SYSTEM OPERATION		312 SEC
PROGRAMMED INJECTION ATTITUDE MANEUVER S/C G & N MONITORING OF MANEUVER		
S-IVB ENGINE CUTOFF	T = 3.01	

T = 2.92

0 5 10 15 20 25

TRAJECTORY DATA - ESTIMATED



GOSS COVERAGE - ESTIMATED

KAUAI

POINT ARGUELLO

DSIF GOLDSTONE

PERTINENT FUNCTIONS

COMMUNICATION & INSTRUMENTATION SYSTEM

NEAR EARTH TELEMETRY

NEAR EARTH 2-WAY VOICE WITH GOSS.

NEAR EARTH 2-WAY DOPPLER TRACKING/RANGING

DATA STORAGE RECORDING

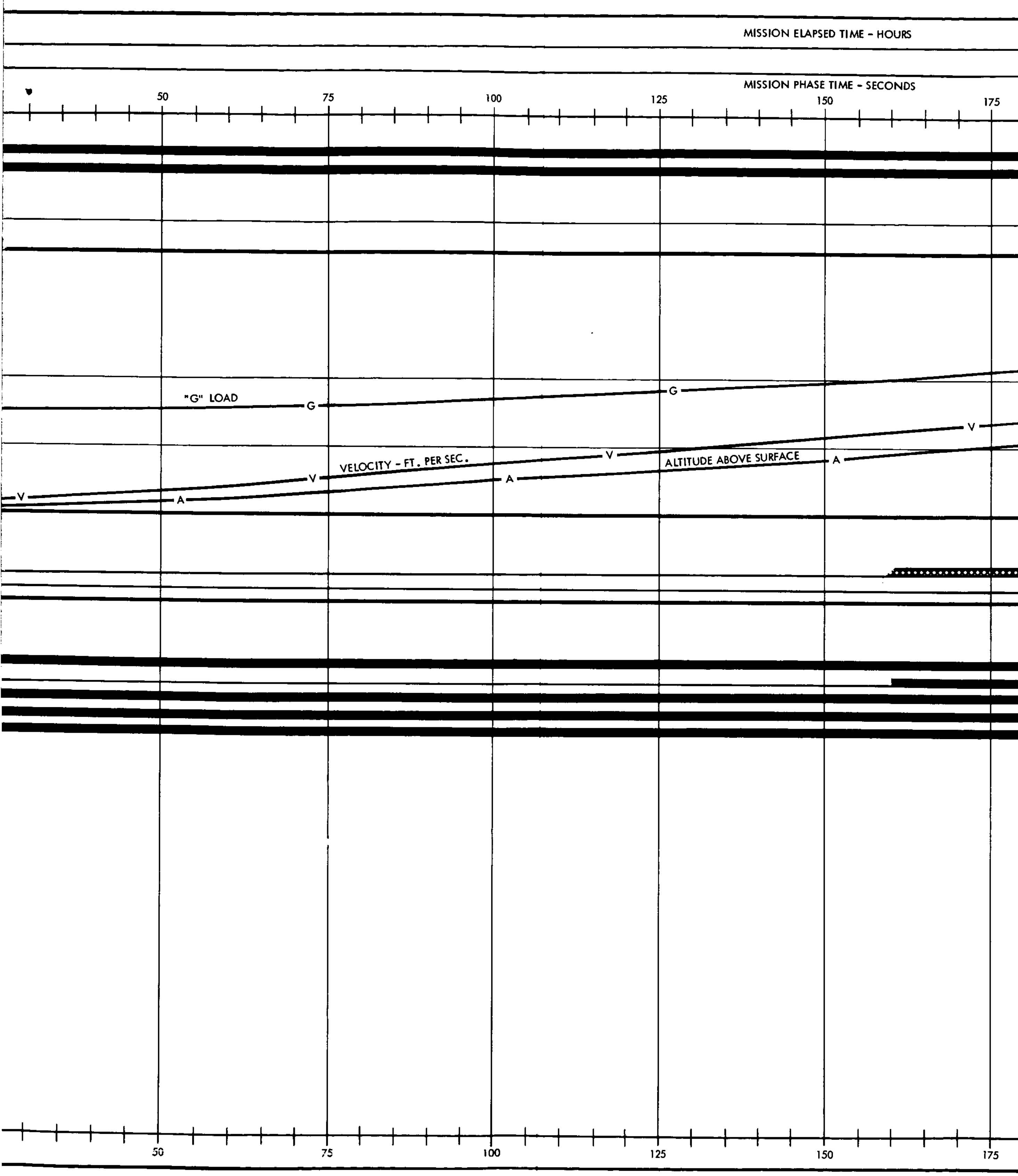
DSIF 2-WAY DOPPLER TRACKING/RANGING

GOSS LEGEND

COMMUNICATIONS.

RADAR & COMMUNICATIONS.

0 5 10 15 20 25


MISSION ELAPSED TIME - HOURS
MISSION PHASE TIME - SECONDS
50
75
100
125
150
175
"G" LOAD
G
G
VELOCITY - FT. PER SEC.
V
V
V
V
ALTITUDE ABOVE SURFACE
A
A
A
A
50
75
100
125
150
175

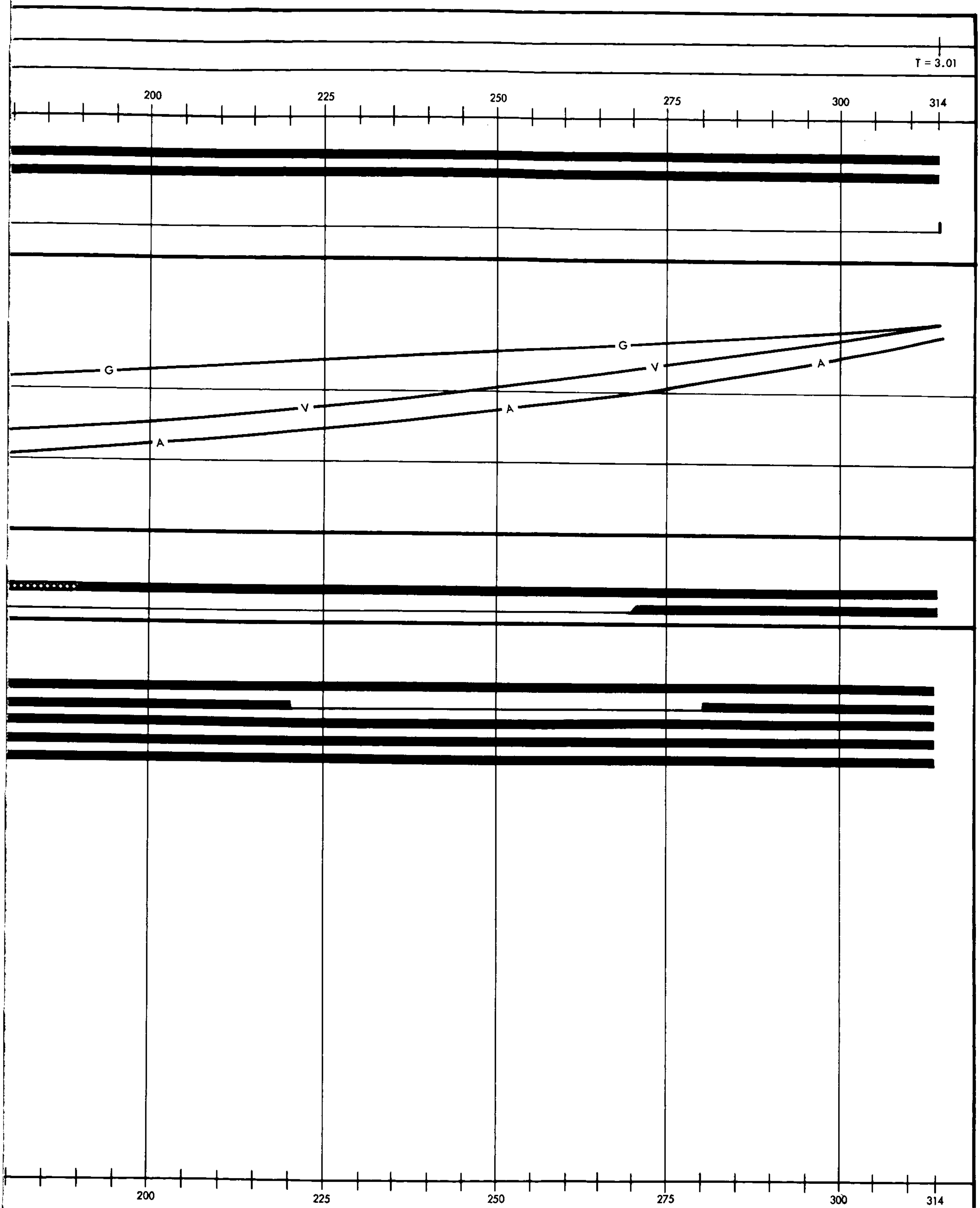


Figure 17. Mission Phase Time Line - Translunar Injection (Sheet 1 of 2)

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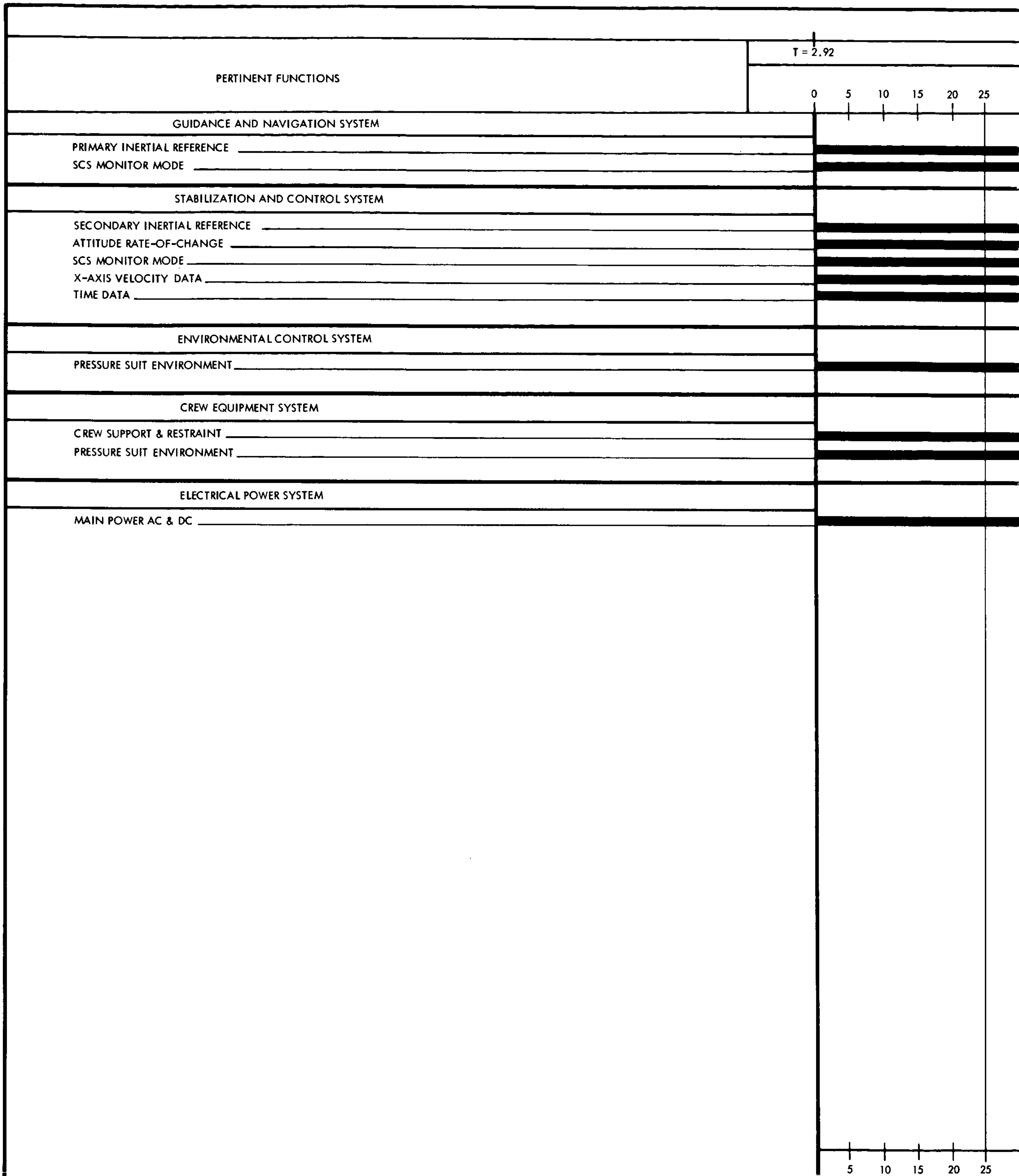
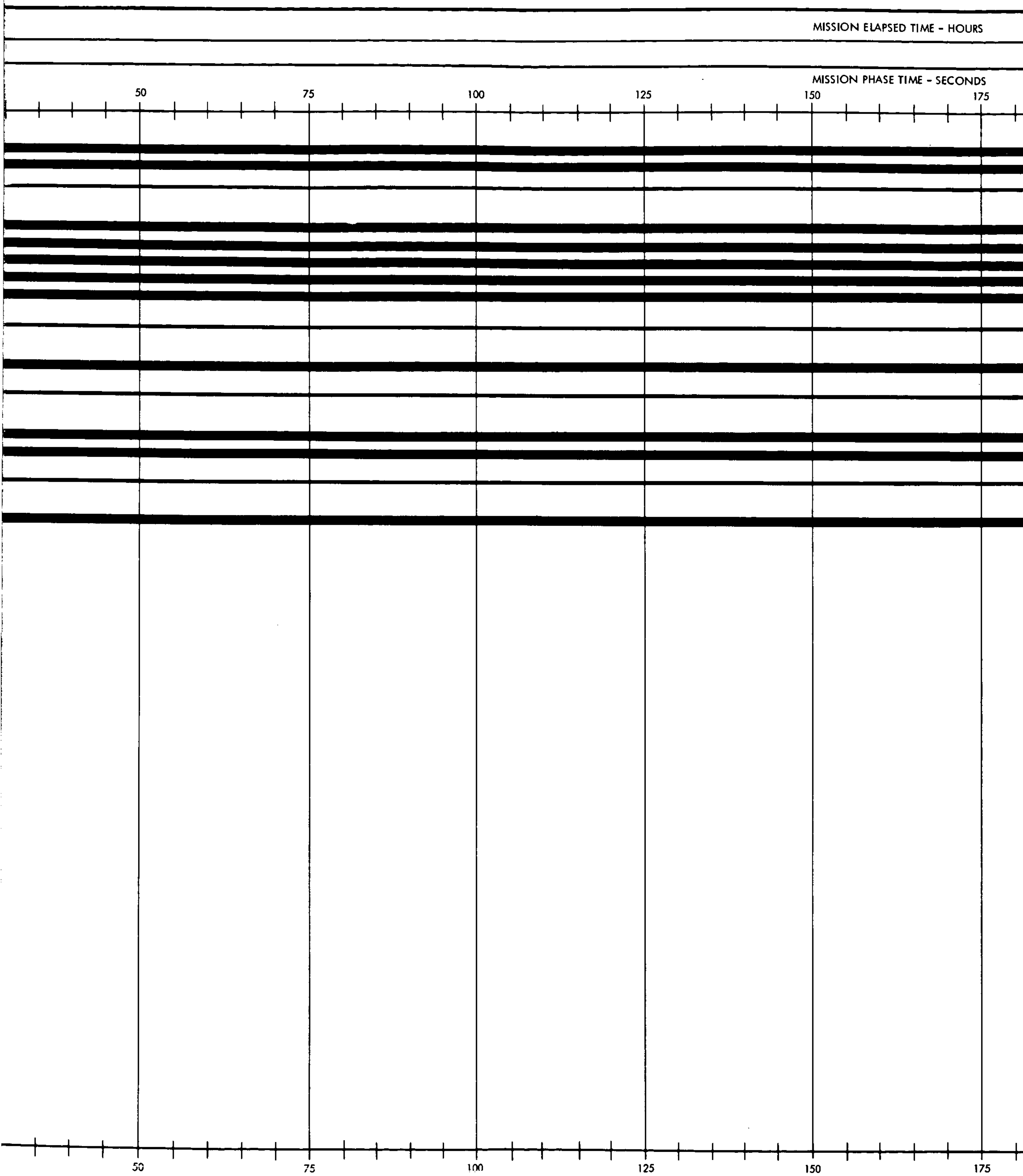




MISSION ELAPSED TIME - HOURS
MISSION PHASE TIME - SECONDS
50
75
100
125
150
175
50
75
100
125
150
175

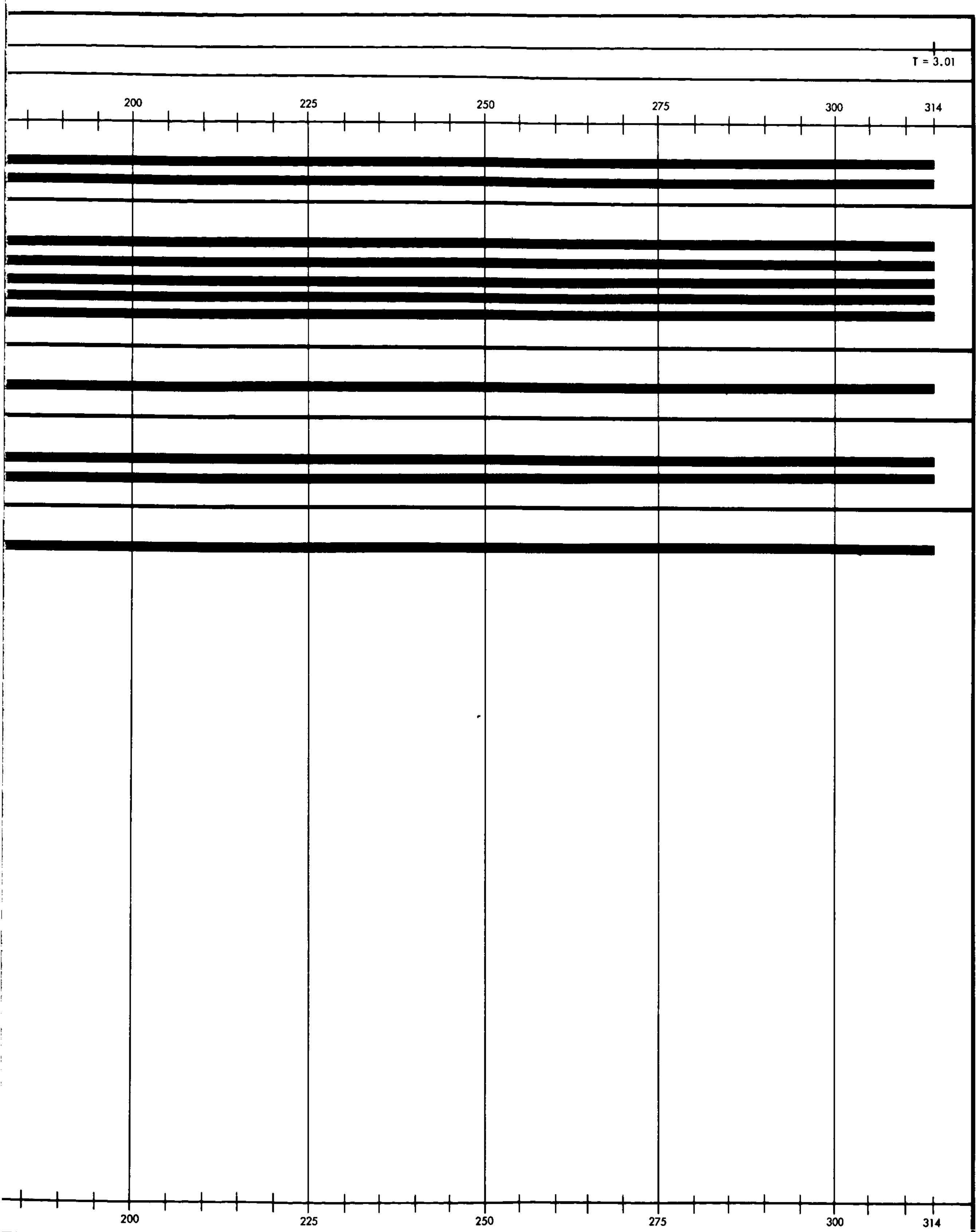


Figure 17. Mission Phase Time Line - Translunar Injection (Sheet 2 of 2)

[redacted]

TRANSLUNAR COAST PHASE

The Translunar Coast Phase begins with S-IVB engine cutoff and ends with S/M Reaction Control System ullage acceleration just prior to lunar orbit injection.

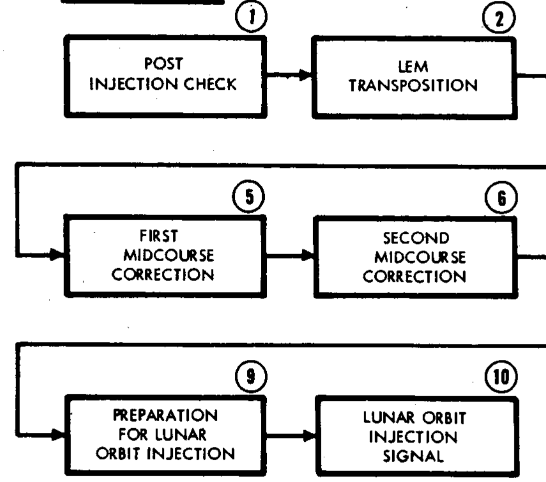
Figure 18 describes the geometry of the Translunar Coast Phase.

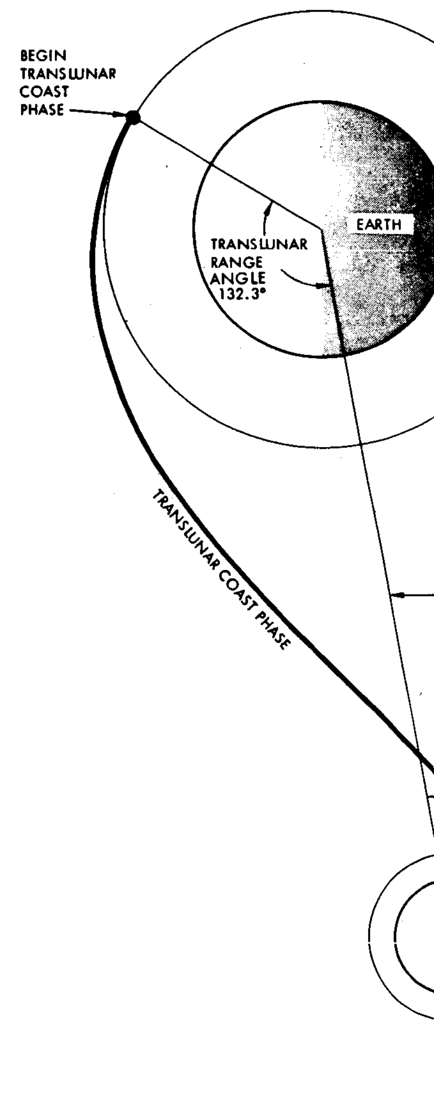
Figure 19 is an earth trace of the Translunar Coast Phase superimposed on a trace for the entire mission.

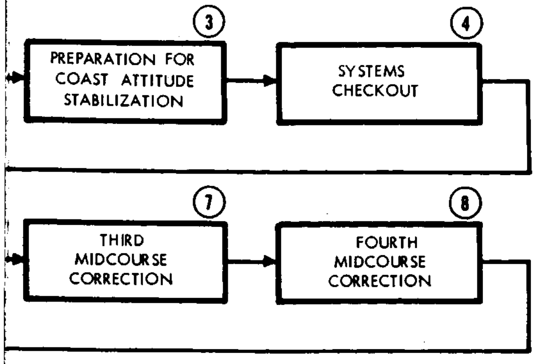
Figure 20 is a two-page time-line delineation of spacecraft system activity during the Translunar Coast Phase.

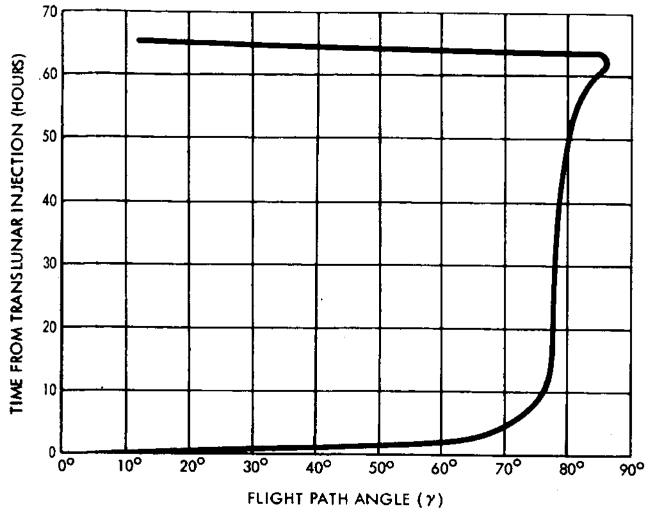
[redacted]

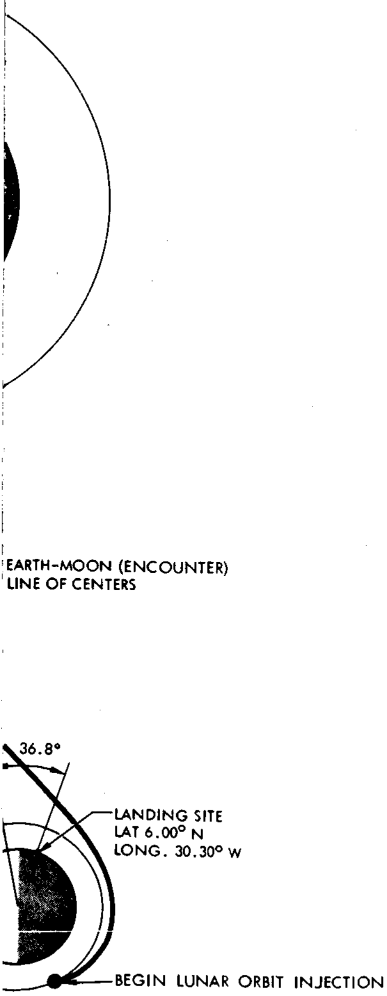
MISSION EVENTS











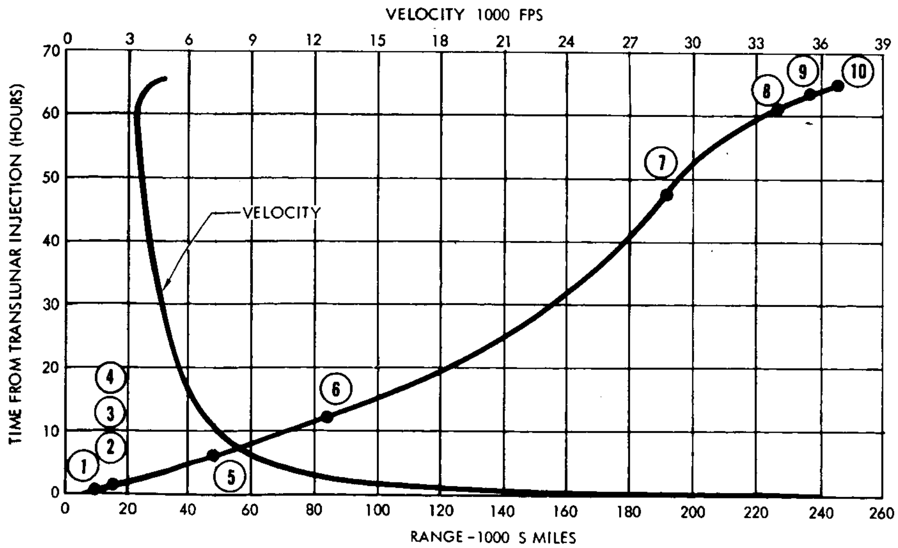
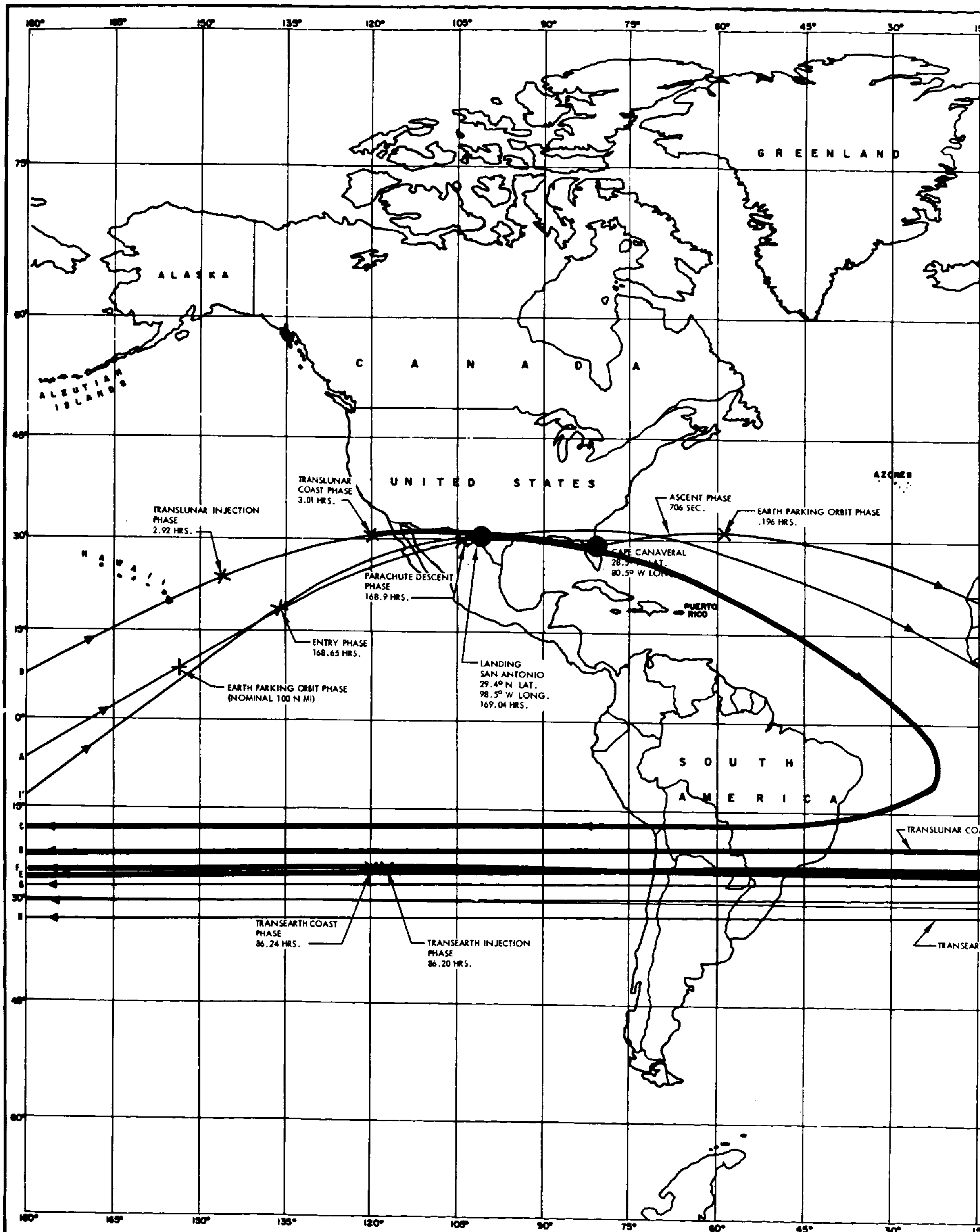


Figure 18. Translunar Coast Phase

[redacted]

[redacted]



GREENLAND
ALASKA
CANADA
ALEUTIAN ISLANDS
UNITED STATES
AZORES
HAWAII
PUERTO RICO
SOUTH AMERICA
TRANSLUNAR COAST PHASE 3.01 HRS.
TRANSLUNAR INJECTION PHASE 2.92 HRS.
ASCENT PHASE 706 SEC.
EARTH PARKING ORBIT PHASE .196 HRS.
CAPE CANAVERAL
PARACHUTE DESCENT PHASE 168.9 HRS.
ENTRY PHASE 168.65 HRS.
EARTH PARKING ORBIT PHASE (NOMINAL 100 N MI)
LANDING SAN ANTONIO 29.4° N LAT. 98.5° W LONG. 169.04 HRS.
TRANSEARTH COAST PHASE 86.24 HRS.
TRANSEARTH INJECTION PHASE 86.20 HRS.
180° 165° 150° 135° 120° 105° 90° 75° 60° 45° 30°
75° 60° 45° 30° 15° 0° 15° 30° 45° 60°

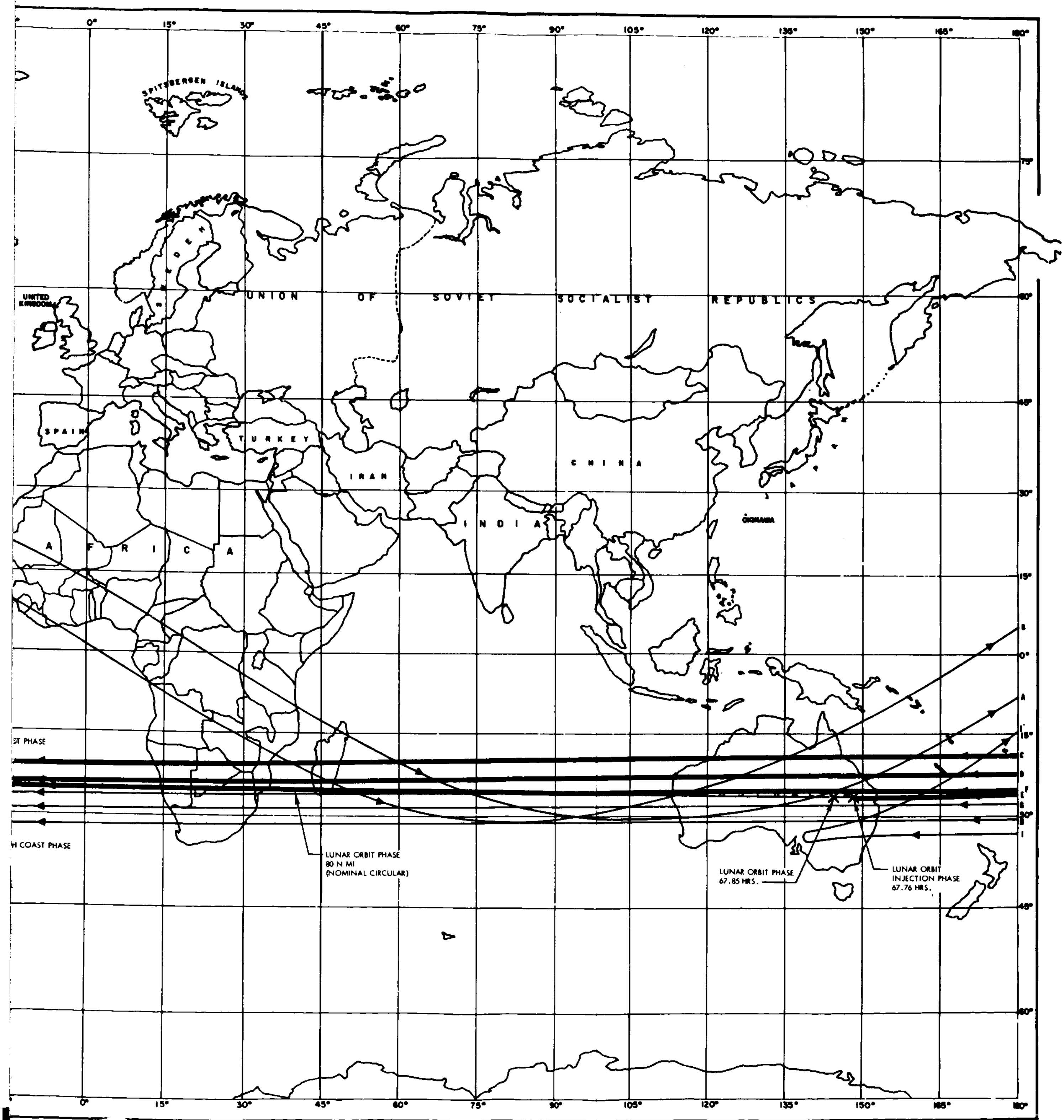


Figure 19. Mission Trajectory Earth Trace-Translunar Coast

[redacted]

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION
S-IVB ENGINE CUTOFF	T = 3.01	
ATTITUDE STABILIZATION - S-IVB		4 HRS
POST INJECTION CHECK & ARRANGEMENT CHECKLIST VERIFICATION OF CONTROL SETTINGS CHECKLIST VERIFICATION OF INSTRUMENT READINGS CREW & EQUIPMENT ARRANGEMENT FOR COAST		15 MIN
LEM TRANSPOSITION LEM ADAPTER SHROUD JETTISON S/C TRANSLATION (APPROX. 150 FORWARD, 180° PITCH, & RETURN) S/C - LEM DOCKING ADAPTER CONE JETTISON & S-IVB SEPARATION	T = 3.51	15 MIN
ATTITUDE STABILIATION		INT.
TRANSLUNAR COAST PREPARATION CREW & EQUIP. ARRANGEMENT FOR TRANSLUNAR COAST	T = 4.01	20 MIN
SYSTEMS CHECKOUT LEM OPERATIONAL VERIFICATION S/C OPERATIONAL VERIFICATION	T = 4.31	50 MIN
1ST MID COURSE CORRECTION MANUAL TRAJECTORY DATA (10 SIGHTINGS - APPROX 1/2 HR. INTERVAL) TRAJECTORY ERROR PARAMETERS COMPUTATION IMU FINE ALIGNMENT (2 SIGHTINGS & GYRO CORRECTION) COUNTDOWN (GIMBAL ACTIVATION, SYST. SET-UP, ORIENTATION) VELOCITY/VECTOR CHANGE POST SPS IMPULSE CHECK (CONTROLS, READOUTS, ARRANGEMENT)	T = 4.31	5 HRS
2ND MIDCOURSE CORRECTION (REFER TO 1ST MIDCOURSE CORRECTION)	T = 10.00	5 HRS
3RD MIDCOURSE CORRECTION (SIMILAR TO 1ST MIDCOURSE CORRECTION EXCEPT THE 10 SIGHTINGS ARE TAKEN AT APPROX. 3 HR. INTERVALS)	T = 20.00	30 HRS
4TH MIDCOURSE CORRECTION (SIMILAR TO 1ST MIDCOURSE CORRECTION EXCEPT THE 10 SIGHTINGS ARE TAKEN AT APPROX. 1 HR. INTERVALS)	T = 55.00	10 HRS
LUNAR ORBIT INJECTION PARAMETERS COMPUTATION COMPUTER PROCESSING OF ONBOARD AND GOSS DATA IMU FINE ALIGNEMENT COUNTDOWN FOR LUNAR ORBIT INJECTION	T = 66.76	1 HR
LUNAR ORBIT INJECTION START SIGNAL	T = 67.76	

GOSS DSIF COVERAGE - ESTIMATED

WOOMERA
JOHANNESBURG
GOLDSTONE

POSITIONAL DATA - ESTIMATED

BEHIND MOON
S/C IN VAN ALLEN RADIATION BELT

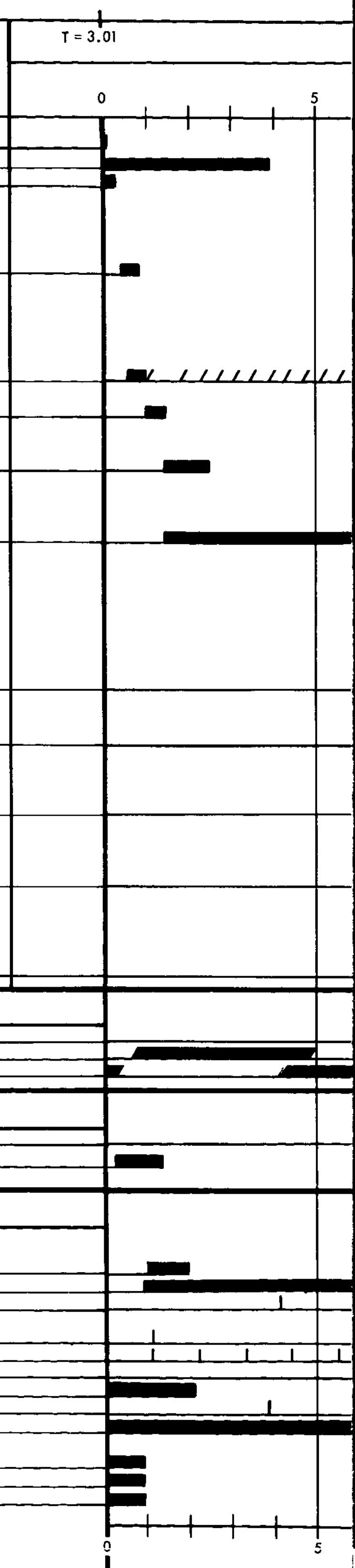
PERTINENT FUNCTIONS

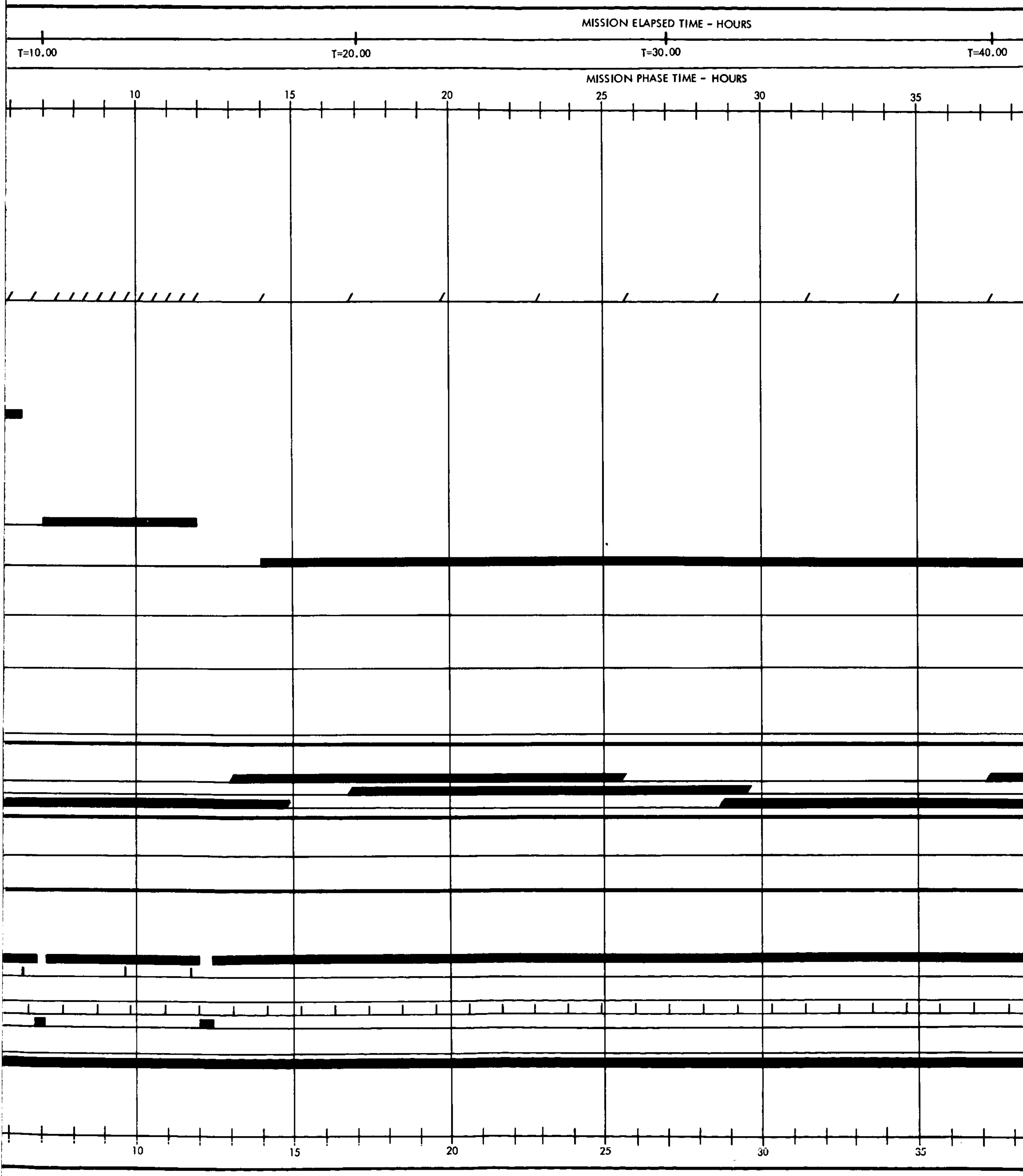
COMMUNICATIONS & INSTRUMENTATION SYSTEM

TWO-WAY VOICE WITH LEM
DSIF NARROW BAND TELEMETRY
DATA STORAGE RECORDING

TWO-WAY VOICE WITH BELT PACKS
DSIF 2-WAY VOICE WITH GOSS
DSIF DATA STORAGE TRANSMISSION
NEAR EARTH 2-WAY DOPPLER TRACKING/RANGING
C/M DSIF TV TRANSMISSION
DSIF 2-WAY DOPPLER TRACKING/RANGING

(MAIN ANTENNA INOPERABLE)
- DSIF 2-WAY VOICE WITH GOSS
- TWO WAY DOPPLER TRACKING/RANGING
- DSIF NARROW BAND TELEMETRY




MISSION ELAPSED TIME - HOURS
T=10.00
T=20.00
T=30.00
T=40.00
MISSION PHASE TIME - HOURS
10
15
20
25
30
35
10
15
20
25
30
35

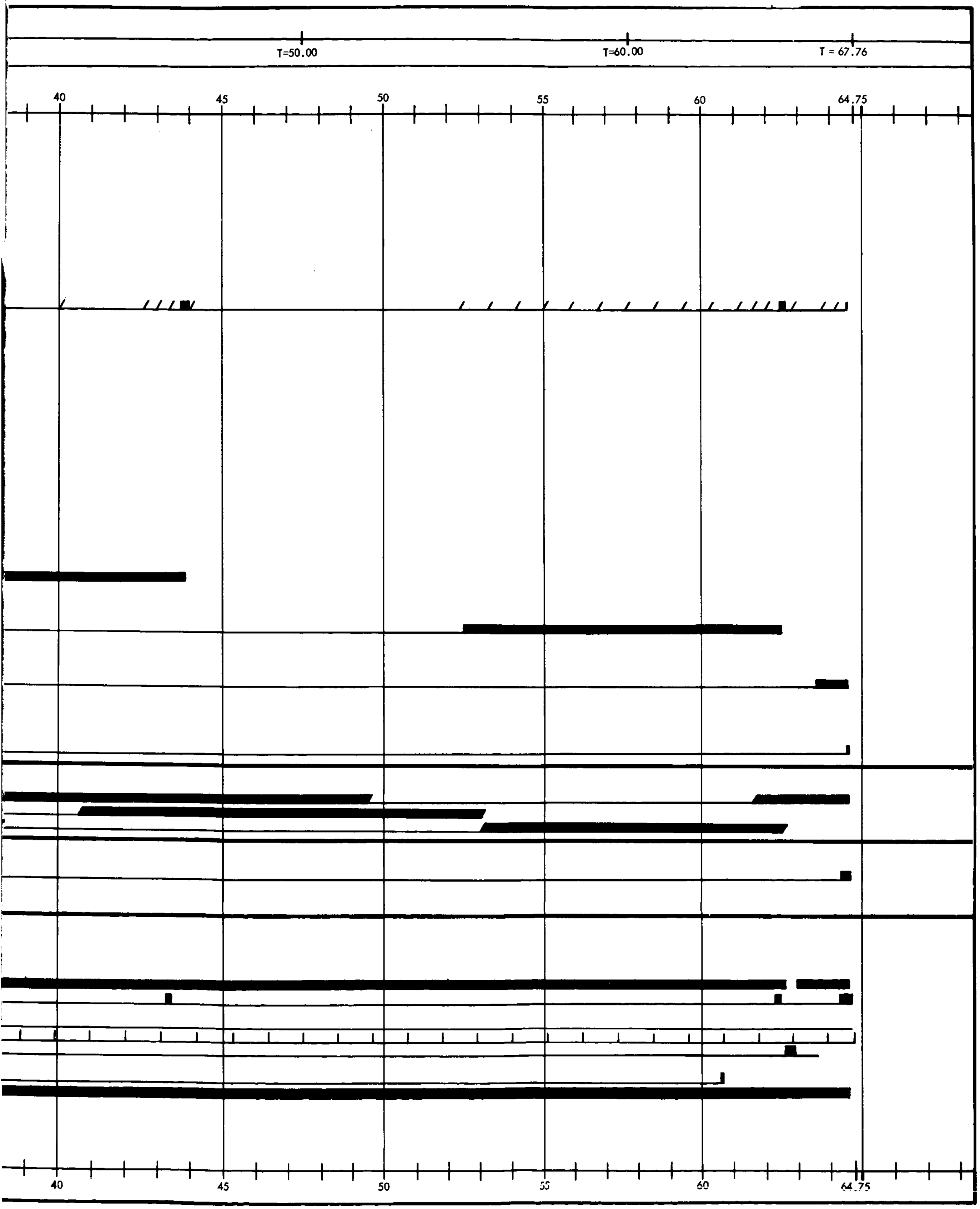


Figure 20. Mission Phase Time Line - Translunar Coast (Sheet 1 of 2)

[redacted]

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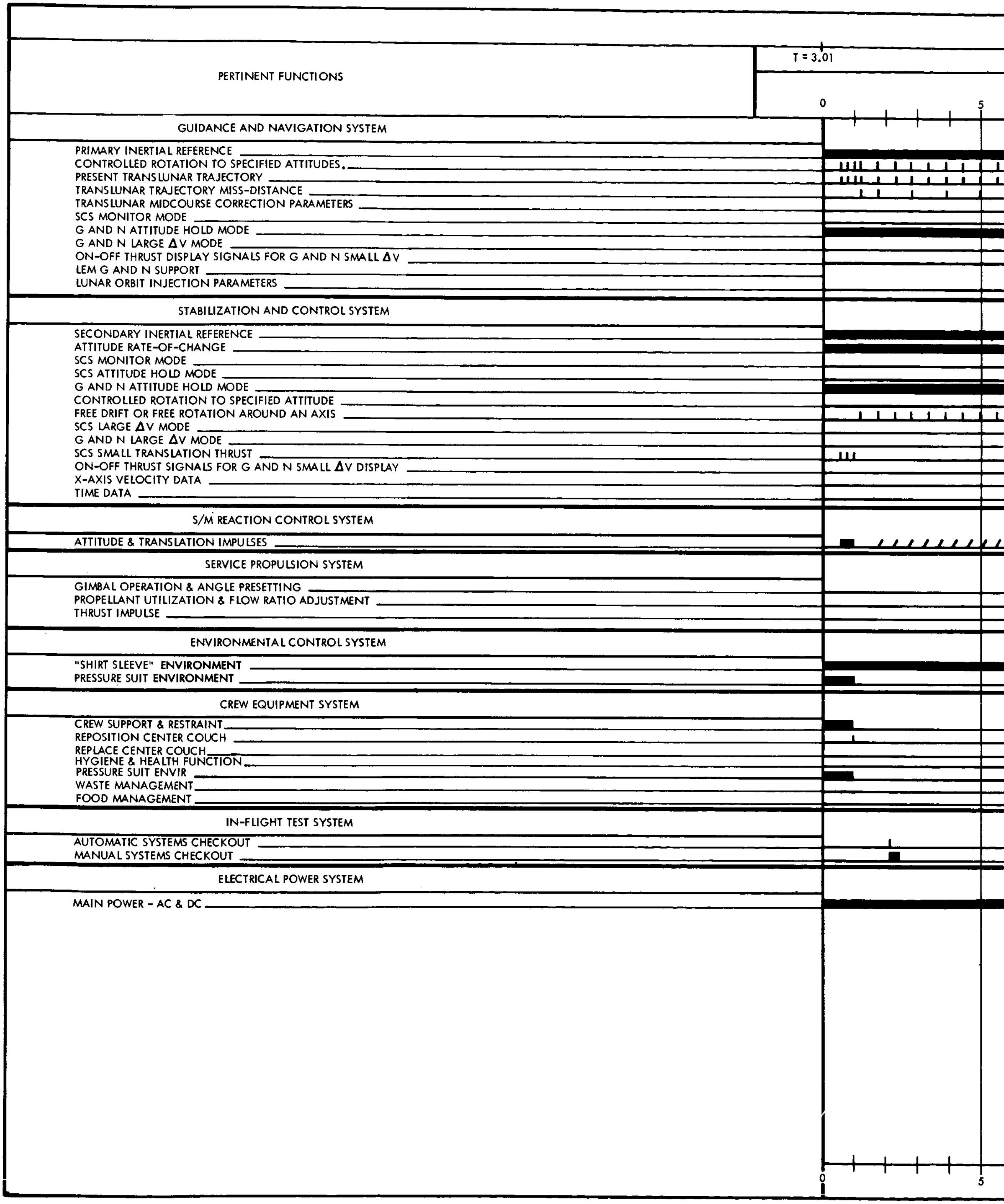
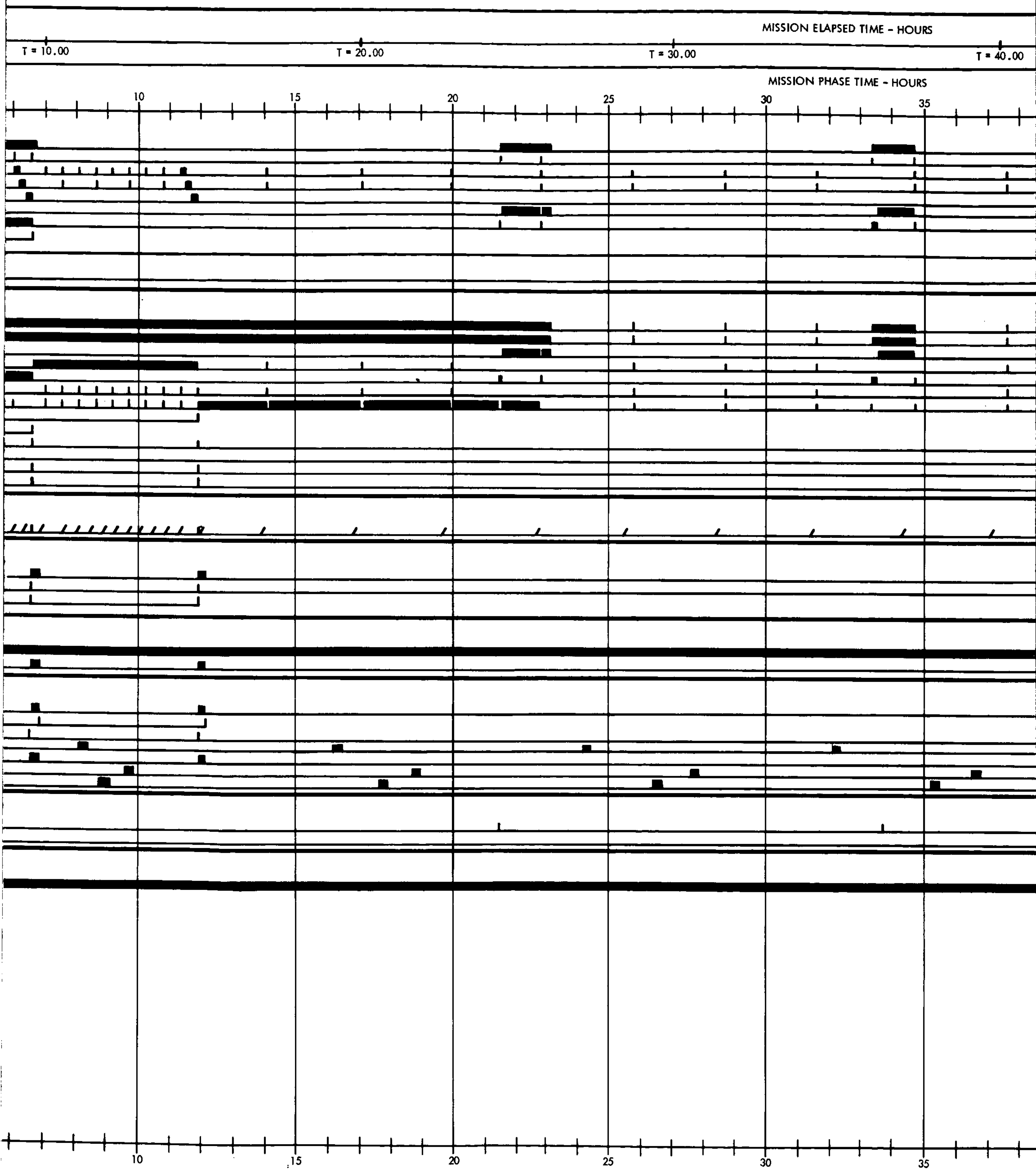



MISSION ELAPSED TIME - HOURS
T = 10.00
T = 20.00
T = 30.00
T = 40.00
MISSION PHASE TIME - HOURS
10
15
20
25
30
35

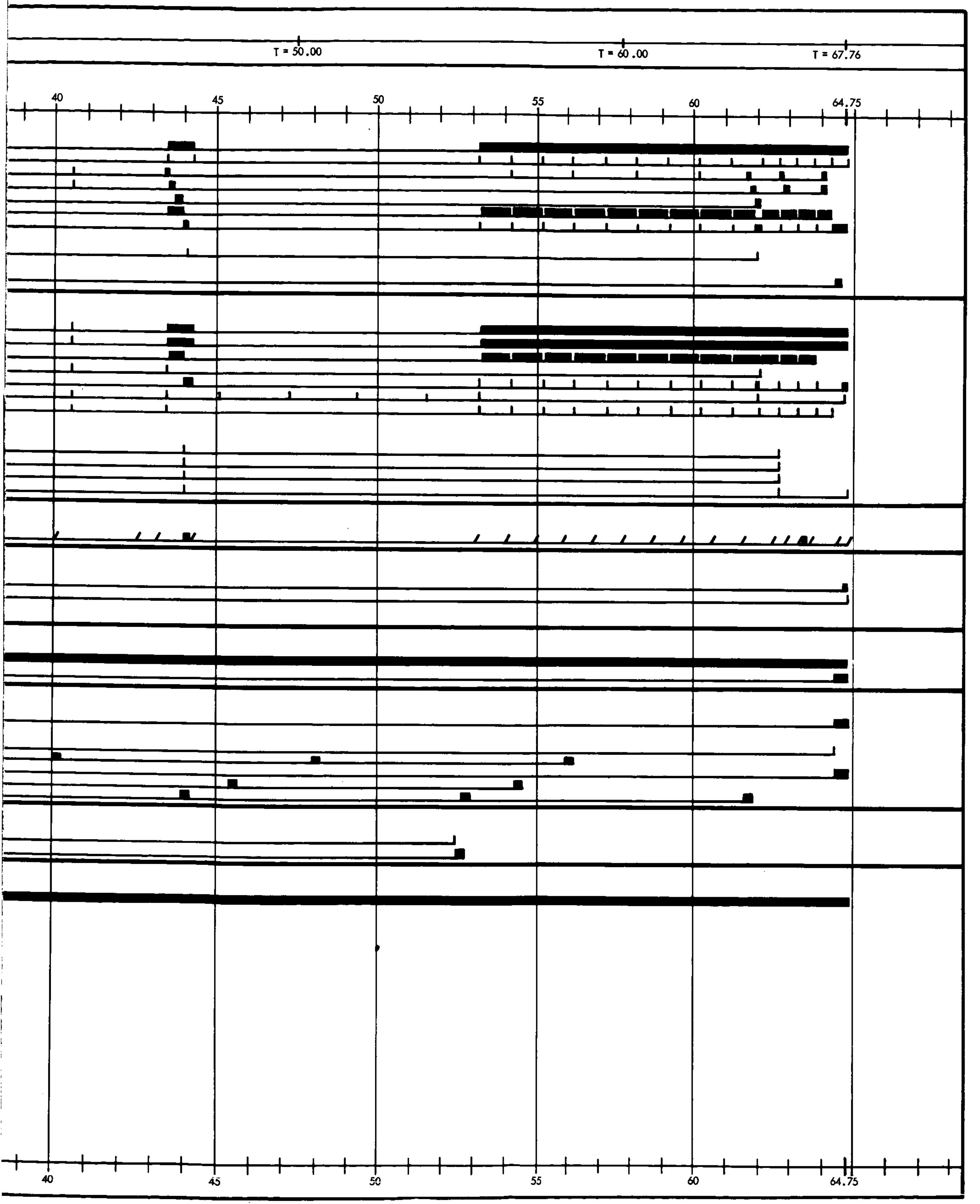


Figure 20. Mission Phase Time Line - Translunar Coast (Sheet 2 of 2)

[redacted]

LUNAR ORBIT INJECTION PHASE

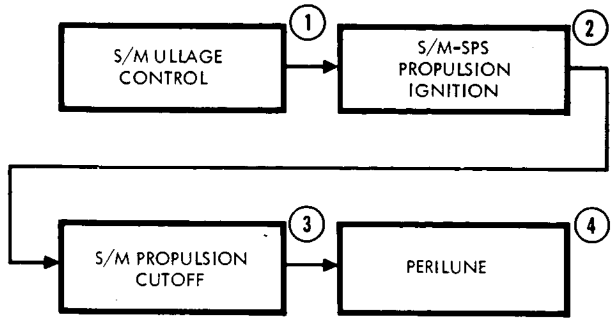
The Lunar Orbit Injection Phase begins with S/M Reaction Control System ullage acceleration and ends with Service Propulsion System cut-off as the spacecraft is injected into an 80 n.mi. lunar orbit.

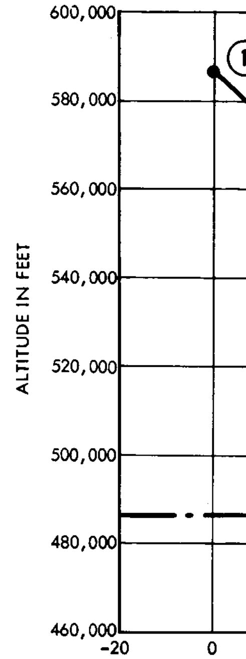
Figure 21 describes the geometry of the Lunar Orbit Injection Phase.

Figure 22 is an earth trace of the Lunar Orbit Injection Phase superimposed on a trace for the entire mission.

Figure 23 is a two-page time-line delineation of spacecraft system activity during the Lunar Orbit Injection Phase.

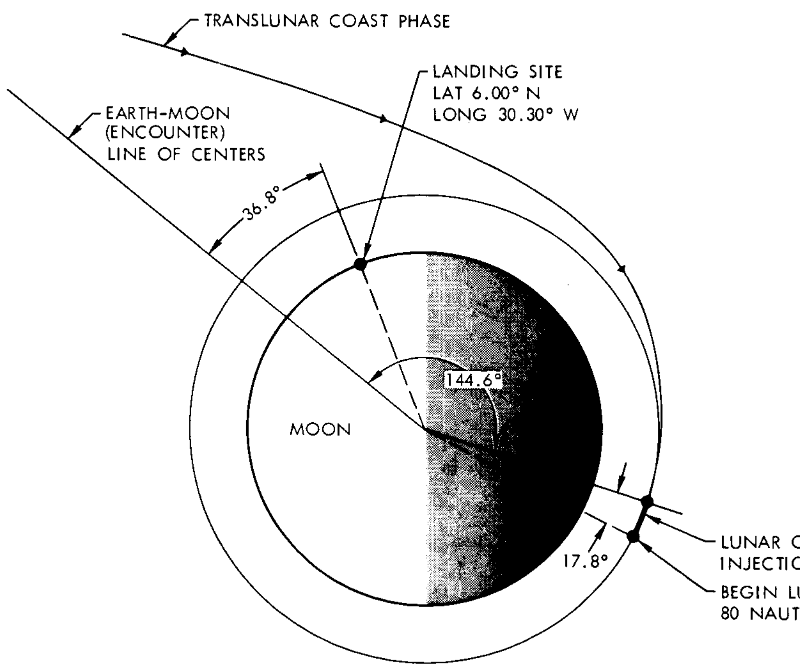
MISSION EVENTS

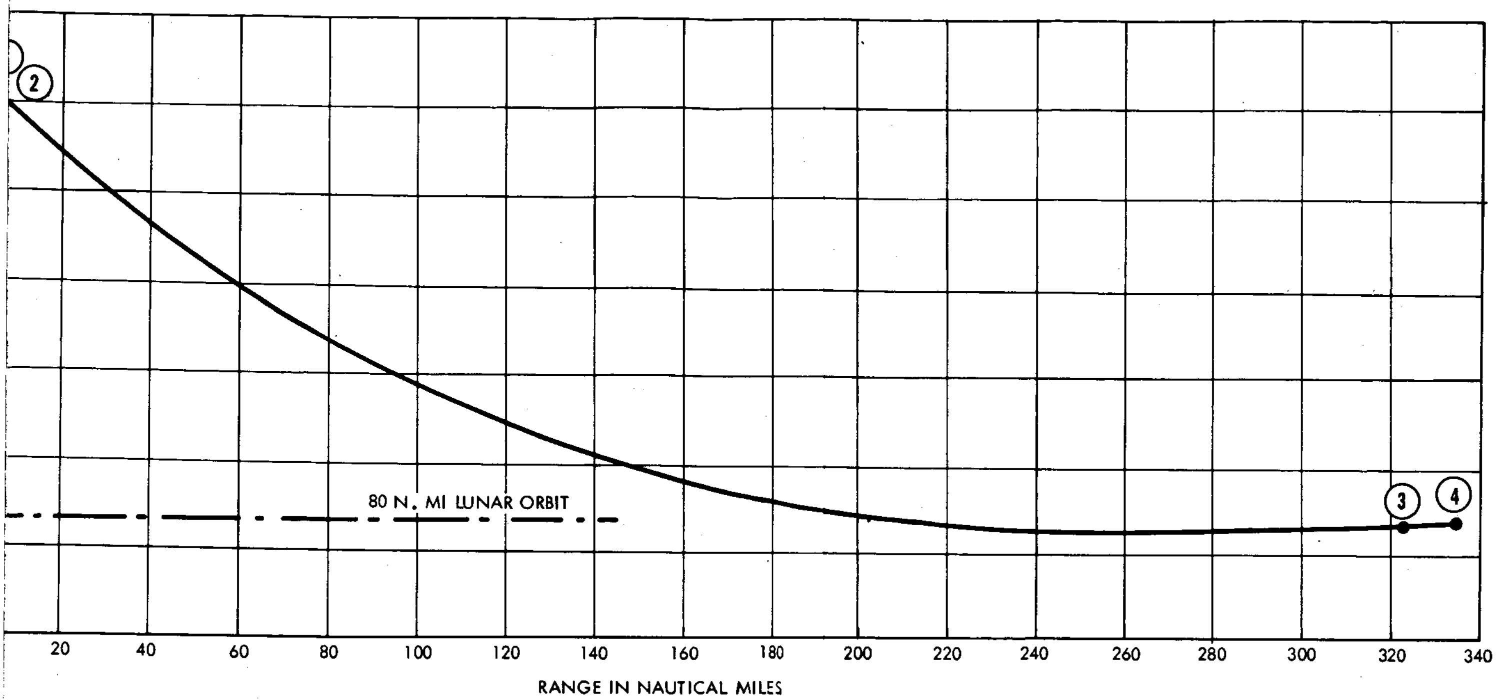




SERVICE MODULE S/M:

ISP = 319.5 SECONDS
THRUST = 21,900 LBS.
T/W_o = .2589
T/W_F = .3506





RBIT
N PHASE
JNAR ORBIT PHASE
CAL MILE ALTITUDE PERILUNE

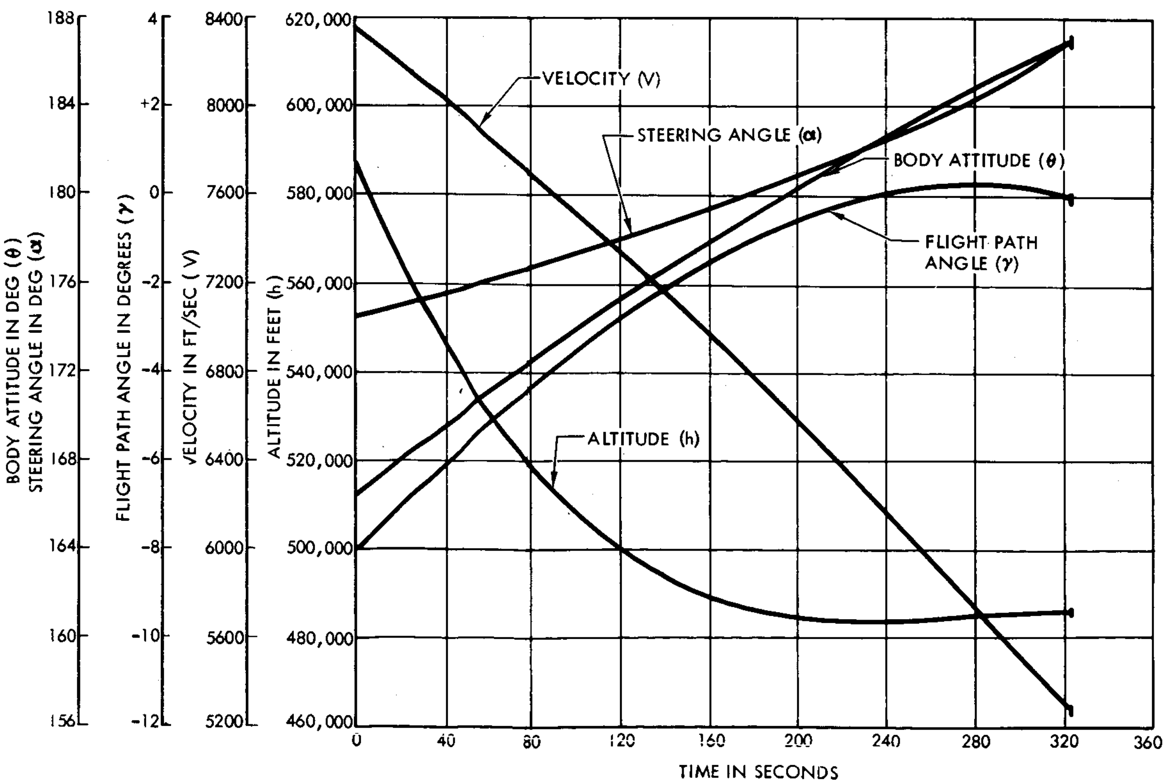
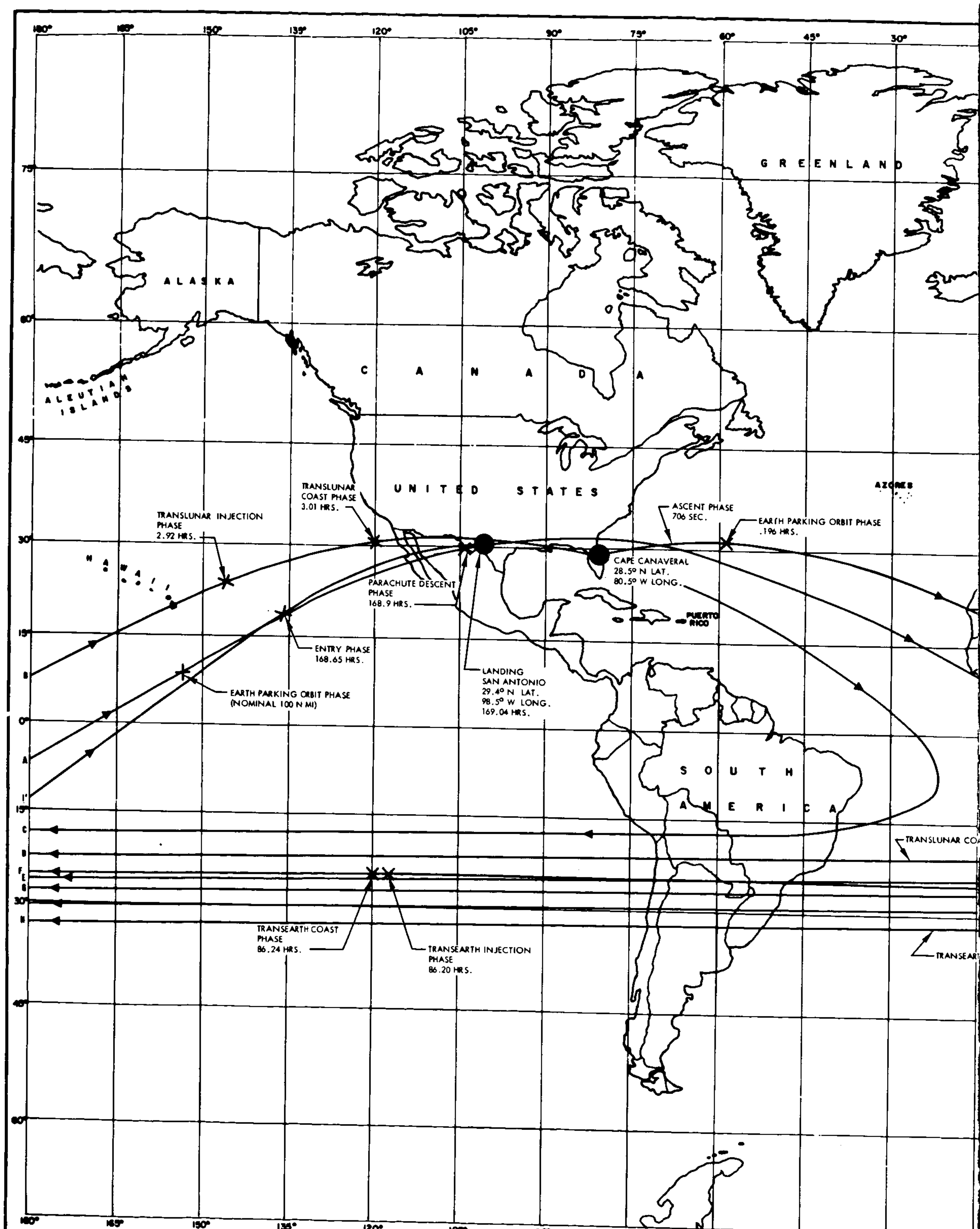


Figure 21. Lunar Orbit Injection Phase

[redacted]

[redacted]



GREENLAND
ALASKA
ALEUTIAN ISLANDS
CANADA
UNITED STATES
AZORES
HAWAII
PUERTO RICO
SOUTH AMERICA
TRANSLUNAR COAST PHASE
3.01 HRS.
TRANSLUNAR INJECTION PHASE
2.92 HRS.
ASCENT PHASE
706 SEC.
EARTH PARKING ORBIT PHASE
.196 HRS.
CAPE CANAVERAL
28.5° N LAT.
80.5° W LONG.
PARACHUTE DESCENT PHASE
168.9 HRS.
ENTRY PHASE
168.65 HRS.
EARTH PARKING ORBIT PHASE
(NOMINAL 100 N MI)
LANDING
SAN ANTONIO
29.4° N LAT.
98.5° W LONG.
169.04 HRS.
TRANSEARTH COAST PHASE
86.24 HRS.
TRANSEARTH INJECTION PHASE
86.20 HRS.
180°
165°
150°
135°
120°
105°
90°
75°
60°
45°
30°
75°
60°
45°
30°
15°
0°
15°
30°
45°
60°

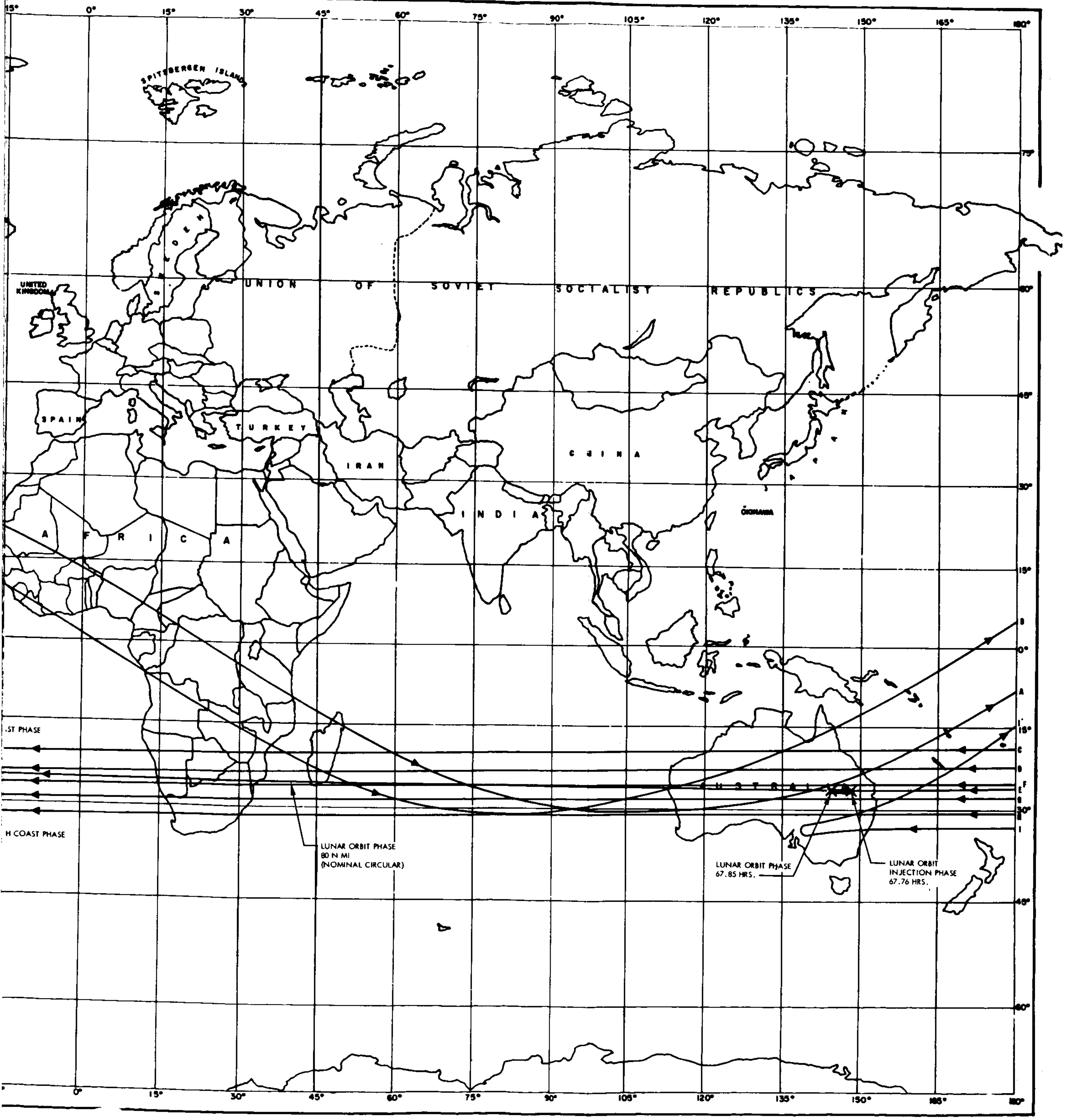


Figure 22. Mission Trajectory Earth Trace-Lunar Orbit Injection

[redacted]

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HRS	EVENT DURATION	T = 67.76 0 5 10 15 20 25
S/M RCS IMPULSE FOR ULLAGE ACCELERATION	T= 67.76	3 SEC	
S/M SPS IGNITION & OPERATION		320 SEC	
G & N PROGRAMMED MANEUVER			
S/M SPS CUTOFF	T = 67.85		

POSITIONAL DATA - ESTIMATED

S/C OVER OPPOSITE SIDE OF MOON FROM EARTH

PERTINENT FUNCTIONS

COMMUNICATION & INSTRUMENTATION SYSTEM

DATA STORAGE RECORDING

0 5 10 15 20

MISSION ELAPSED TIME ~ HOURS

MISSION PHASE TIME ~ SECONDS

50 100 150

30 50 100 150

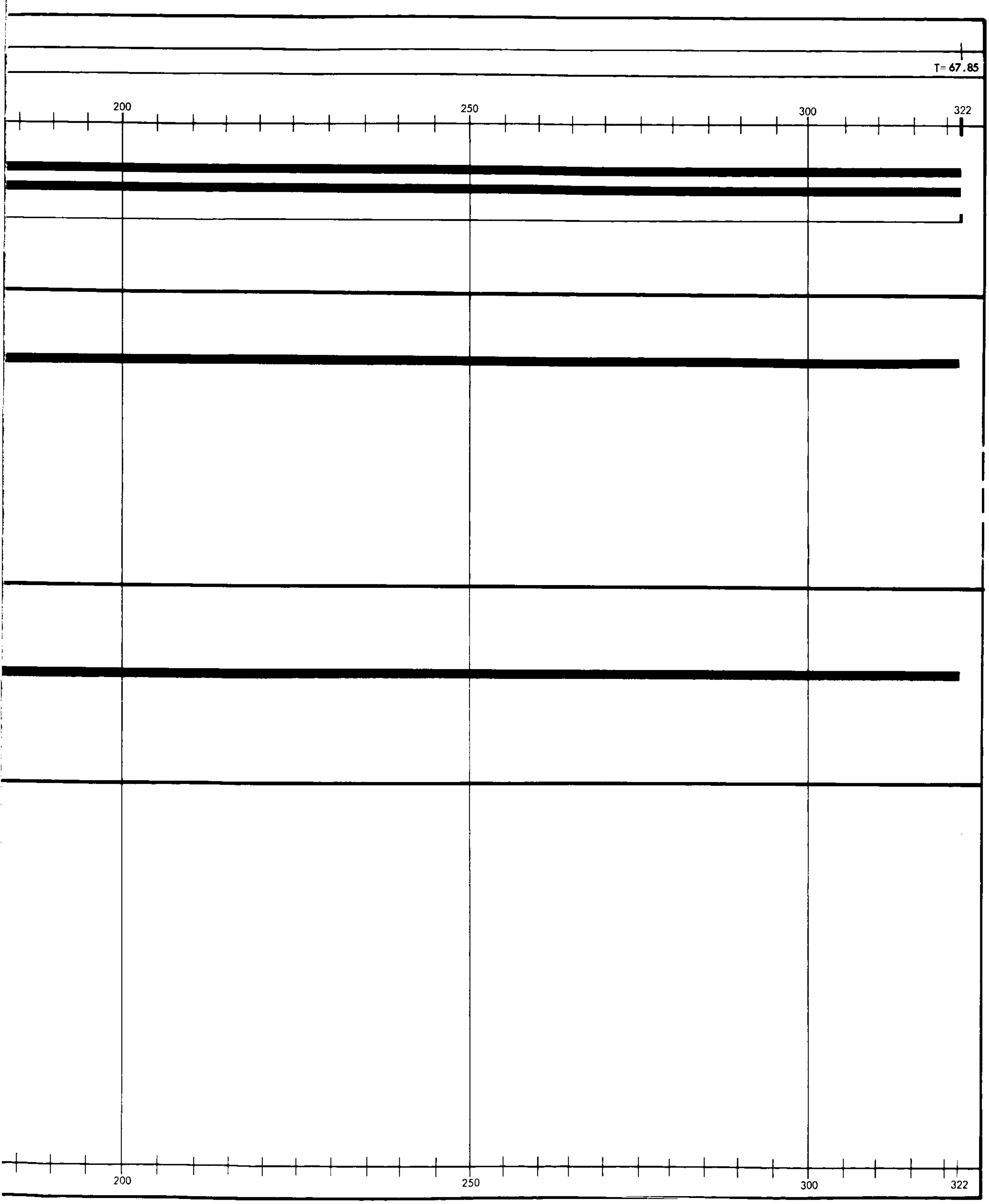
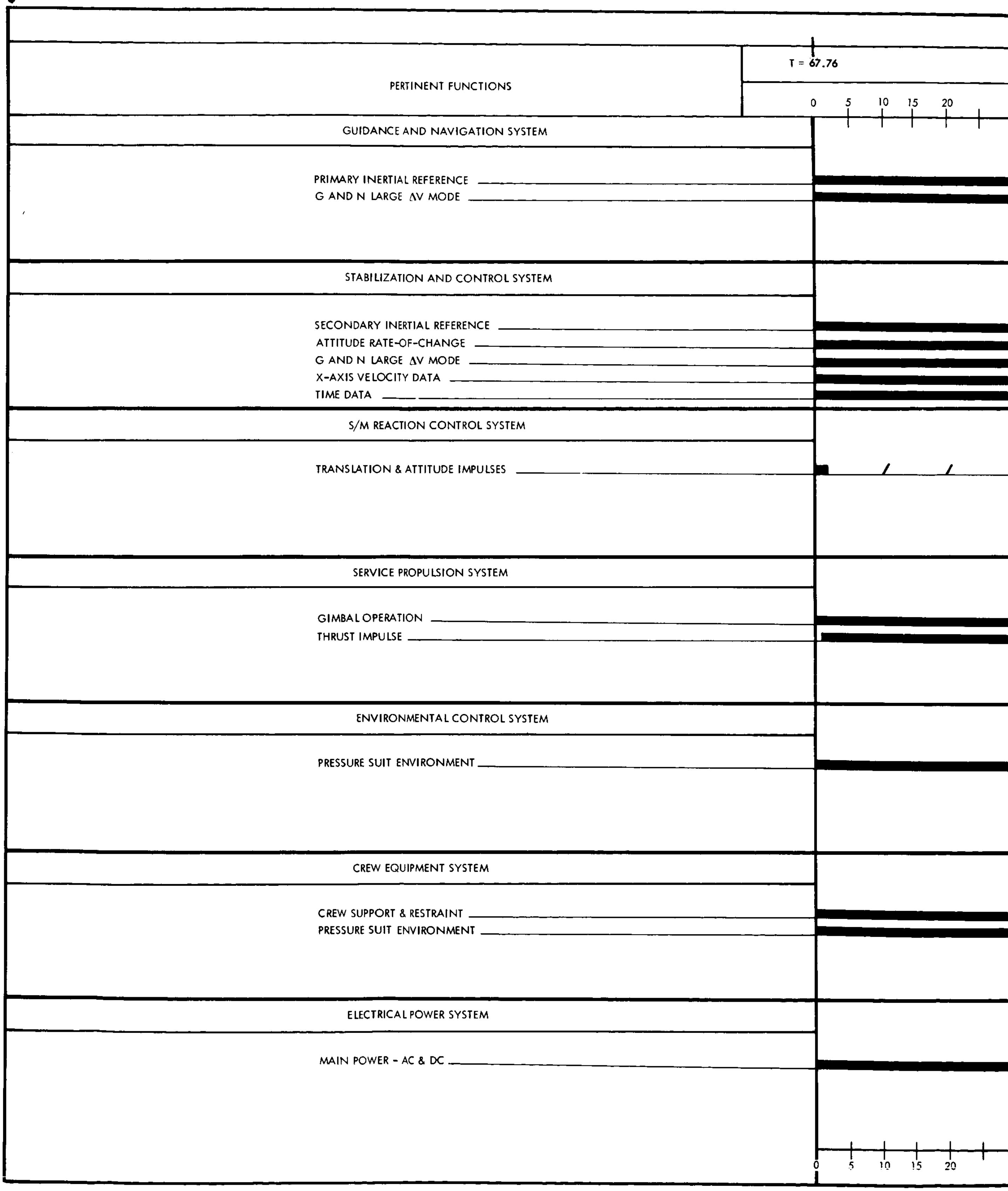


Figure 23. Mission Phase Time Line - Lunar Orbit Injection (Sheet 1 of 2)

[redacted]


PERTINENT FUNCTIONS
T = 67.76
0
5
10
15
20
GUIDANCE AND NAVIGATION SYSTEM
PRIMARY INERTIAL REFERENCE
G AND N LARGE ΔV MODE
STABILIZATION AND CONTROL SYSTEM
SECONDARY INERTIAL REFERENCE
ATTITUDE RATE-OF-CHANGE
G AND N LARGE ΔV MODE
X-AXIS VELOCITY DATA
TIME DATA
S/M REACTION CONTROL SYSTEM
TRANSLATION & ATTITUDE IMPULSES
SERVICE PROPULSION SYSTEM
GIMBAL OPERATION
THRUST IMPULSE
ENVIRONMENTAL CONTROL SYSTEM
PRESSURE SUIT ENVIRONMENT
CREW EQUIPMENT SYSTEM
CREW SUPPORT & RESTRAINT
PRESSURE SUIT ENVIRONMENT
ELECTRICAL POWER SYSTEM
MAIN POWER - AC & DC
0
5
10
15
20

MISSION ELAPSED TIME - HOURS

MISSION PHASE TIME - SECONDS

30 50 100 150

30 50 100 150

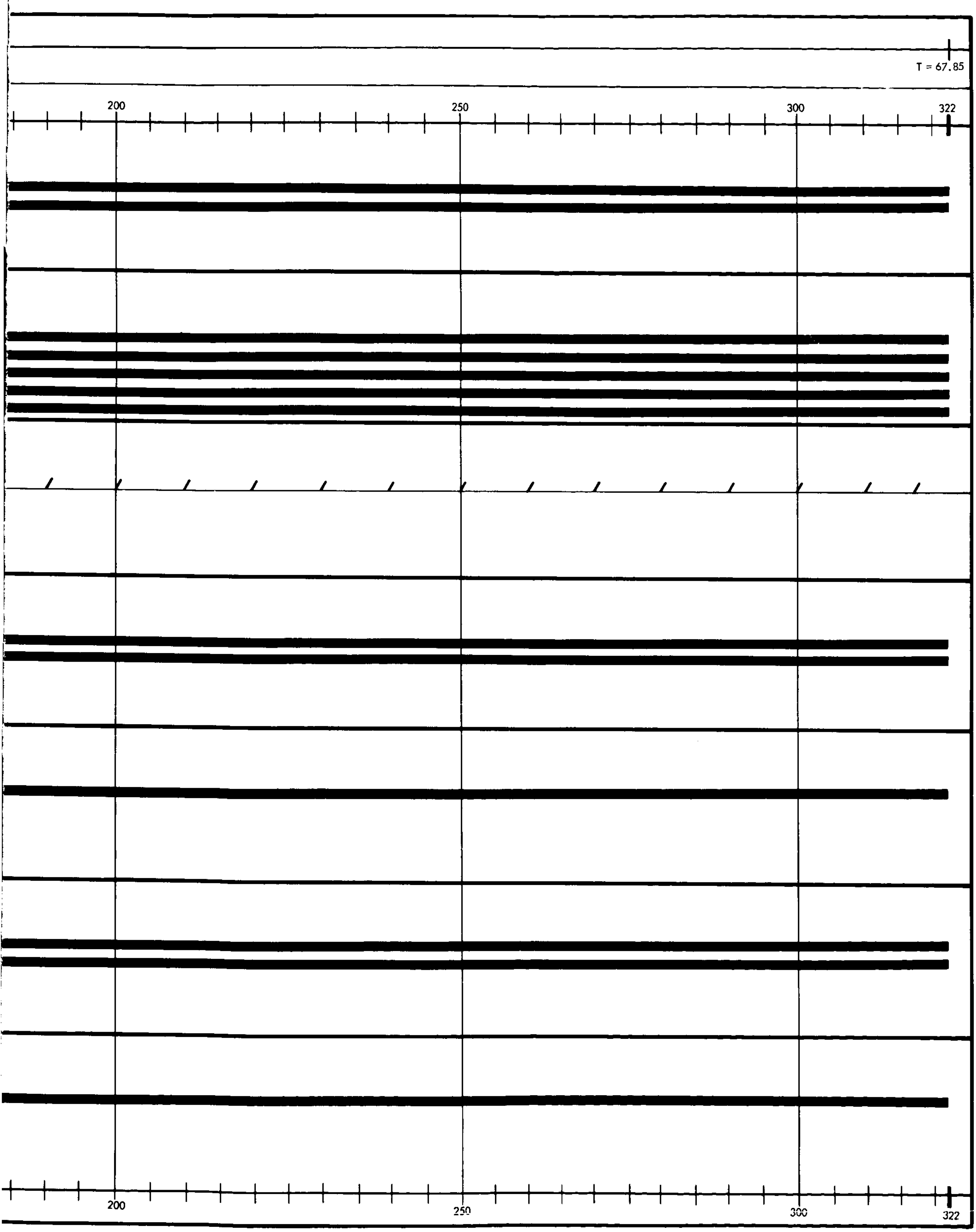


Figure 23. Mission Phase Time Line - Lunar Orbit Injection (Sheet 2 of 2)

[redacted]

[redacted]

LUNAR ORBIT PHASE

(Prior to LEM Separation)

The Lunar Orbit Phase (Prior to LEM Separation) begins with Service Propulsion System cut-off as the spacecraft is injected into lunar orbit. The phase ends with LEM propulsion system ignition.

Figure 24 describes the geometry of this phase.

Figure 25 is an earth trace of this phase and the subsequent two phases superimposed on a trace for the entire mission.

Figure 26 is a two-page time-line delineation of spacecraft system activity during this phase.

 [redacted]

[redacted]

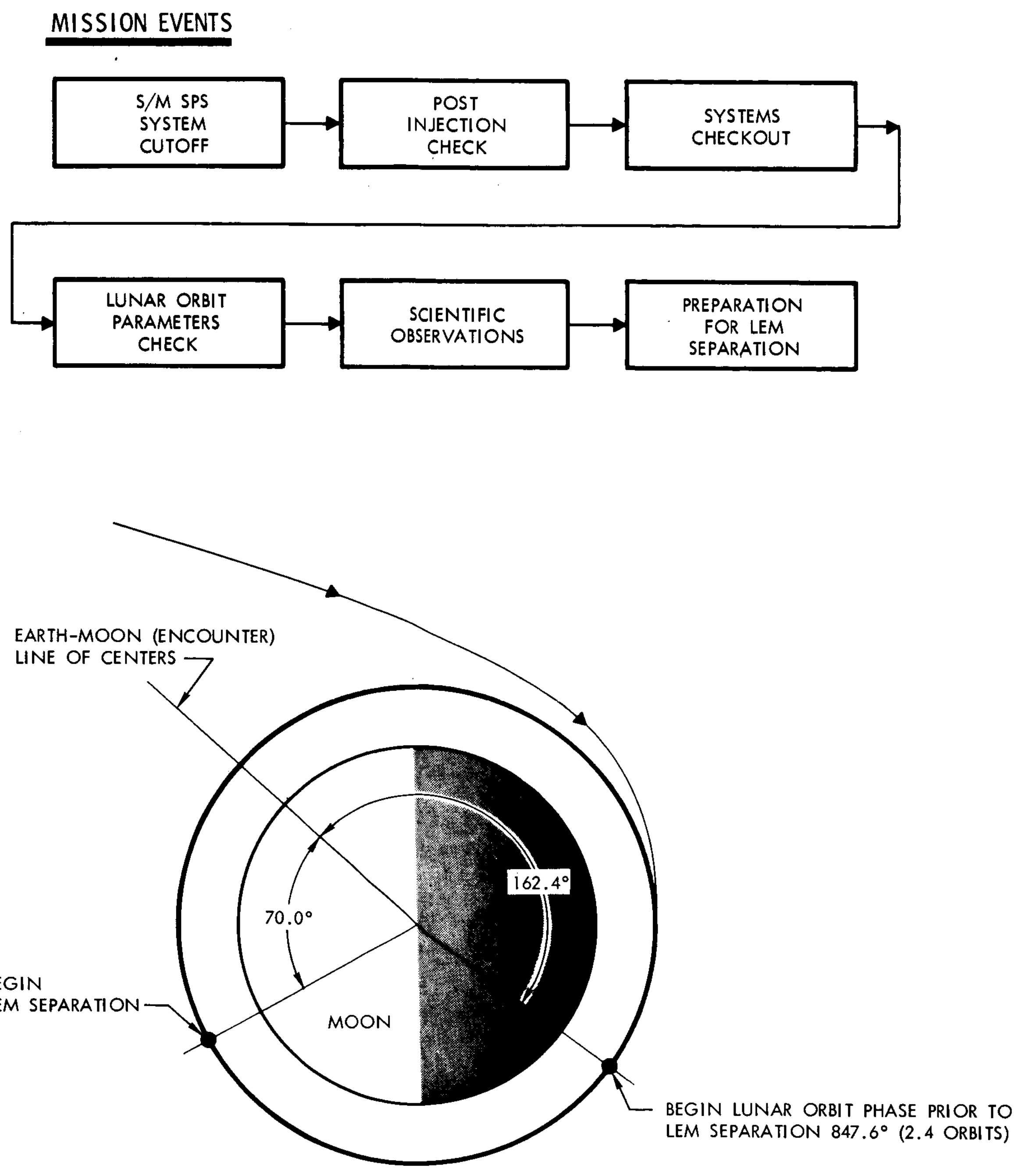


Figure 24. Lunar Orbit Phase

[redacted]

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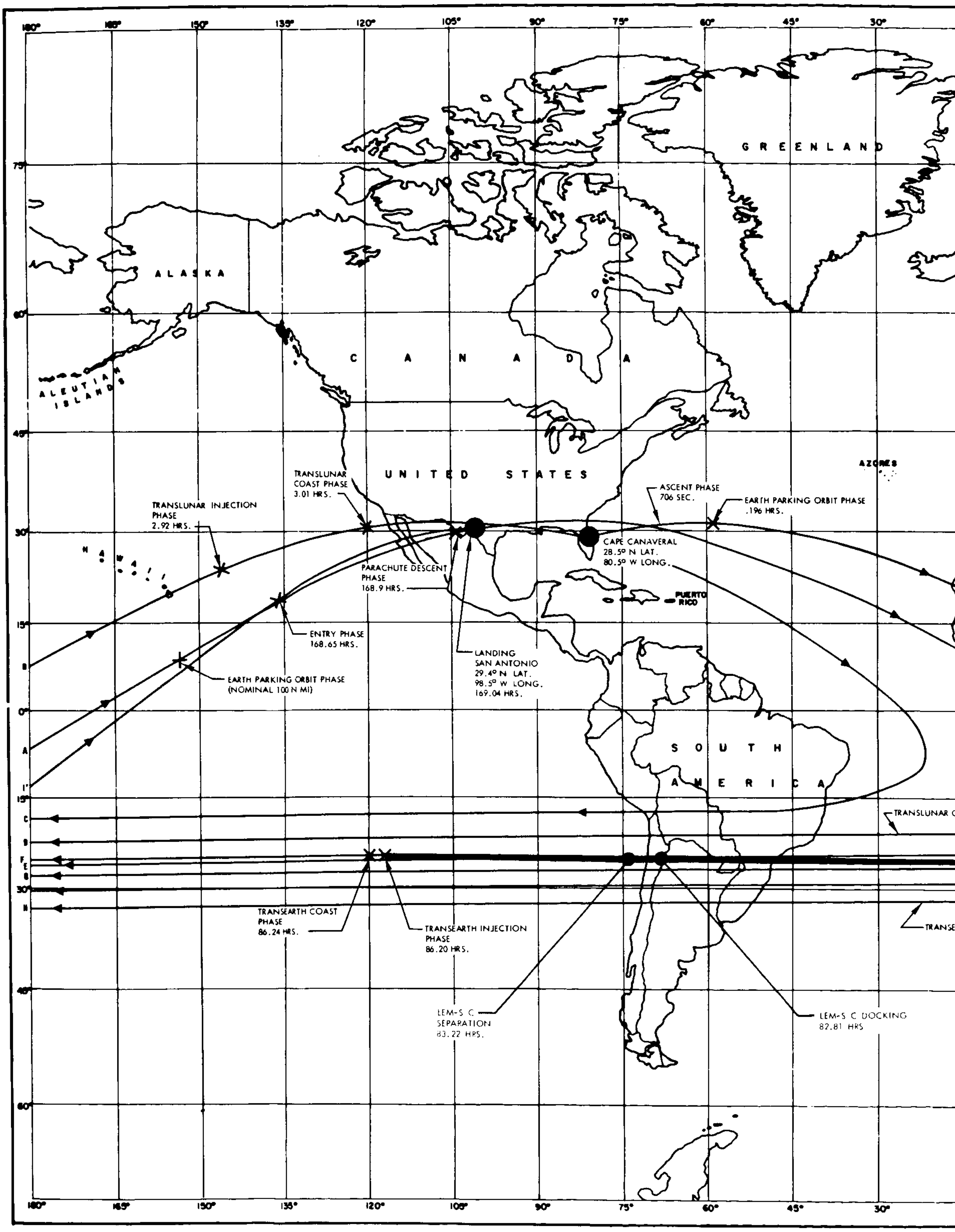


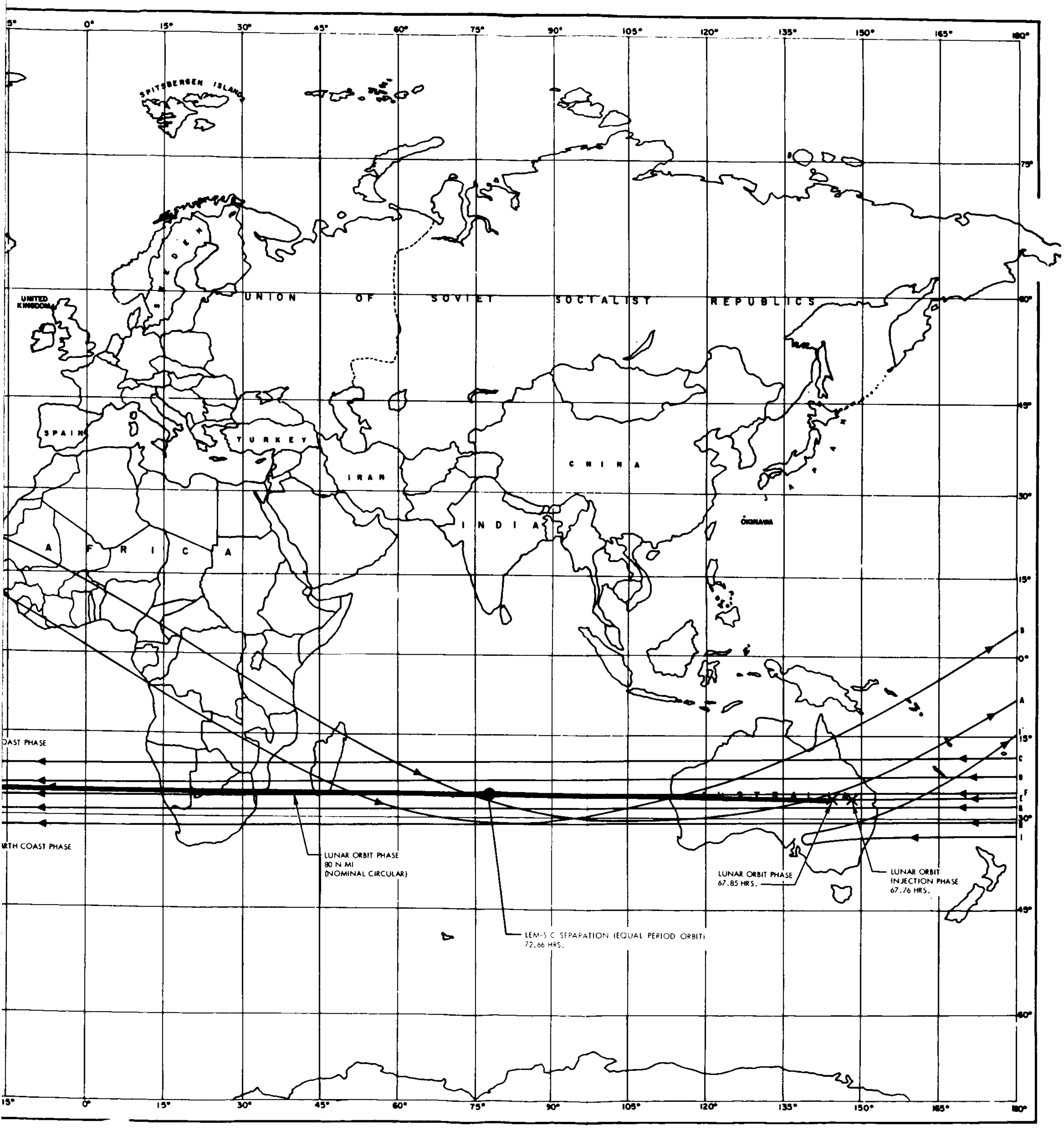


Figure 25. Mission Trajectory Earth Trace-Lunar Orbit

[redacted]

[redacted]

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION	T = 67.85 (0 5 10 15 20 25)
S/M PROPULSION CUTOFF	T = 67.85		
STABILIZATION			
POST INJECTION CHECK & LUNAR ORBIT PREPARATIONS		15 MIN	
CHECKLIST VERIFICATION OF CONTROL SETTINGS			
CHECKLIST VERIFICATION OF INSTRUMENT READINGS			
BIOLOGICAL & RADIATION CHECK			
CREW & EQUIPMENT ARRANGEMENT			
SYSTEMS CHECKOUT/VERIFICATION		15 MIN	
LUNAR ORBIT PARAMETERS CHECK/CORRECTION		180 MIN	
OPTICAL TRACKING OF ORBIT PARAMETERS			
LUNAR ORBIT CORRECTION PARAMETERS			
FINE ALIGN IMU			
COUNTDOWN FOR VELOCITY/VECTOR CHANGE			
VELOCITY/VECTOR CHANGE			
POST SPS IMPULSE CHECK, CREW & EQUIP. ARRANGEMENT			
VERIFICATION OF ORBIT CHANGES			
SCIENTIFIC OBSERVATIONS			
LEM SEPARATION PREPARATION		165 MIN	
SURVEY OF LANDING AREA			
LEM ELLIPTICAL APPROACH PARAMETERS COMPUTATION			
LEM SYSTEMS CHECKOUT			
LEM COUNTDOWN			
START LEM SEPARATION	T = 72.66		

GOSS DSIF COVERAGE - ESTIMATED
GOLDSTONE DSIF
JOHANNESBURG
WOOMERA

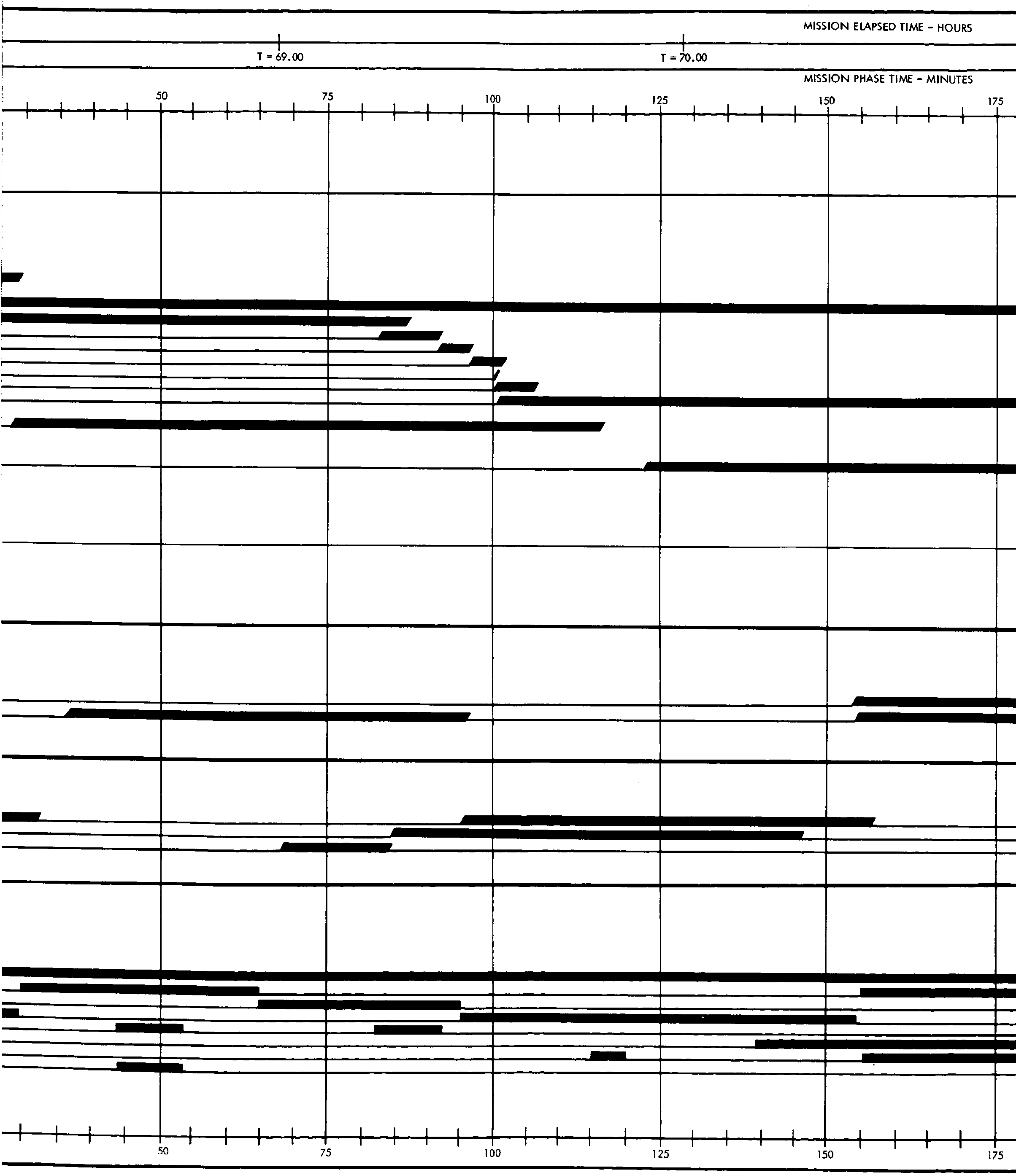
POSITIONAL DATA - ESTIMATED
OPPOSITE SIDE OF MOON
OVER LUNAR SHADOW
S/C IN LINE OF SIGHT WITH LANDING SITE

PERTINENT FUNCTIONS

COMMUNICATIONS & INSTRUMENTATION SYSTEM

- DSIF 2-WAY DOPPLER TRACKING/RANGING
- DSIF NARROW BAND TELEMETRY
- DSIF DATA STORAGE TRANSMISSION
- DATA STORAGE RECORDING
- DSIF 2-WAY VOICE WITH GOSS
- TWO-WAY VOICE WITH LEM
- DSIF 2-WAY VOICE RELAY TO GOSS
- C/M DSIF TV TRANSMISSION

0 5 10 15 20 25


MISSION ELAPSED TIME - HOURS
T = 69.00
T = 70.00
MISSION PHASE TIME - MINUTES
50
75
100
125
150
175

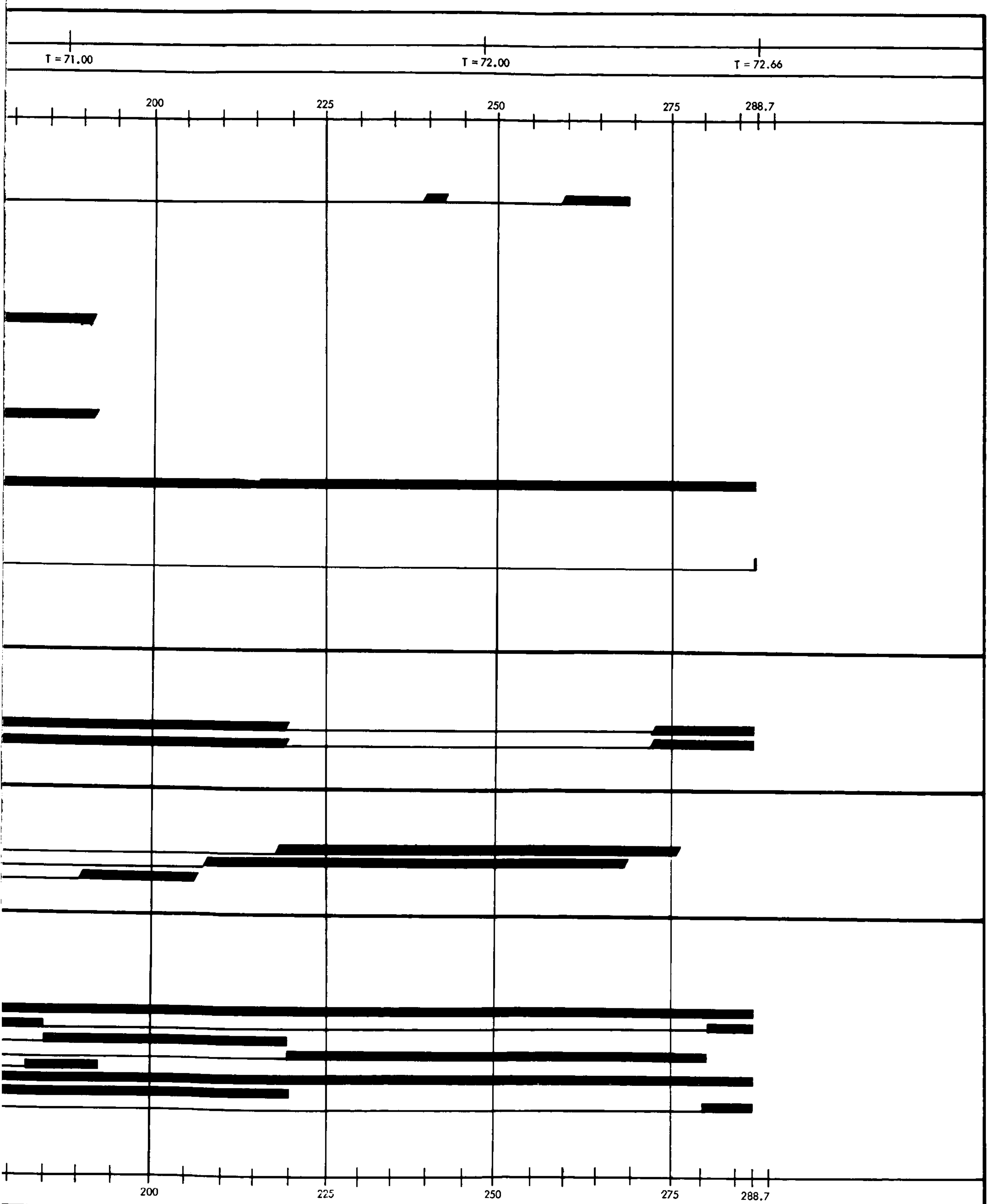
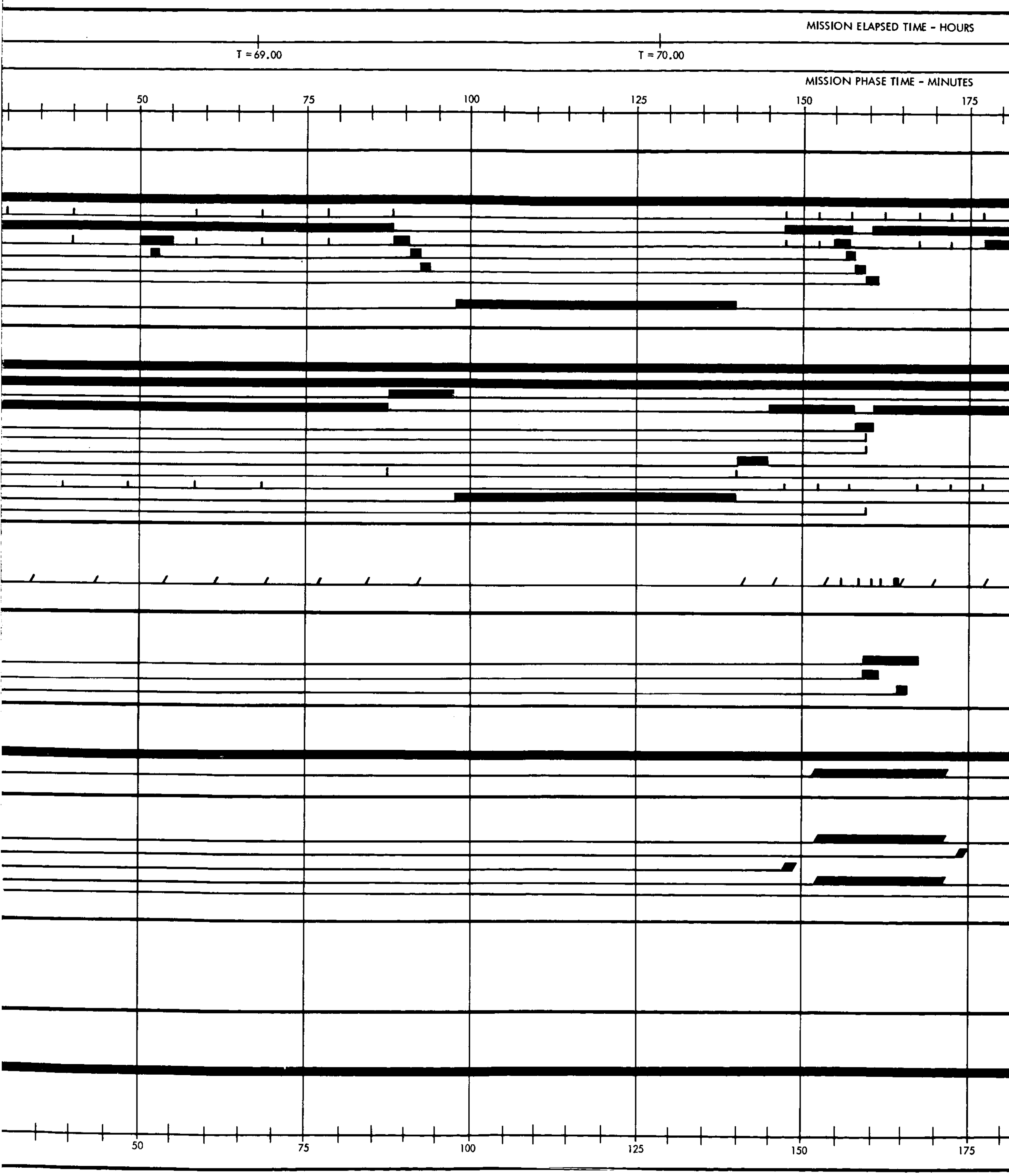


Figure 26. Mission Phase Time Line-Lunar Orbit (Prior to LEM Separation) (Sheet 1 of 2)

[redacted]

PERTINENT FUNCTIONS	T = 67.85 T = 68.00 0 5 10 15 20 25
GUIDANCE AND NAVIGATION SYSTEM	
PRIMARY INERTIAL REFERENCE	
CONTROLLED ROTATION TO SPECIFIED ATTITUDES	
G AND N ATTITUDE HOLD MODE	
LUNAR ORBIT AND EPHEMERIDES	
TRAJECTORY MISS DISTANCE	
MID-COURSE CORRECTION PARAMETERS	
G AND N LARGE Δ V MODE	
LEM G AND N SUPPORT	
SCS MONITOR MODE	
STABILIZATION AND CONTROL SYSTEM	
SECONDARY INERTIAL REFERENCE	
ATTITUDE RATE-OF-CHANGE	
SCS ATTITUDE HOLD MODE	
G AND N ATTITUDE HOLD MODE	
G AND N LARGE Δ V MODE	
X-AXIS VELOCITY DATA	
TIME DATA	
SCS LOCAL VERTICAL MODE	
CONTROLLED ROTATION TO SPECIFIED ATTITUDES	
FREE DRIFT OR ROTATION AROUND AN AXIS	
SCS MONITOR MODE	
SCS SMALL TRANSLATION THRUST	
S/M REACTION CONTROL SYSTEM	
ATTITUDE & TRANSLATION IMPULSES.	
SERVICE PROPULSION SYSTEM	
GIMBAL OPERATION & ANGLE PRESETTING.	
PROPELLANT UTILIZATION & FLOW RATIO ADJUSTMENT.	
THRUST IMPULSE.	
ENVIRONMENTAL CONTROL SYSTEM	
"SHIRT SLEEVE" ENVIRONMENT	
PRESSURE SUIT ENVIRONMENT	
CREW EQUIPMENT SYSTEM	
CREW SUPPORT & RESTRAINT	
REPOSITION CENTER COUCH	
REPLACE CENTER COUCH	
PRESSURE SUIT ENVIRONMENT	
INDIVIDUAL O_2 SUPPLIES - (BACK PACKS)	
IN-FLIGHT TEST SYSTEM	
CRITICAL SYSTEMS CHECKOUT	
ELECTRICAL POWER SYSTEM	
MAIN POWER AC & DC	
	0 5 10 15 20 25


MISSION ELAPSED TIME - HOURS
T = 69.00
T = 70.00
MISSION PHASE TIME - MINUTES
50
75
100
125
150
175

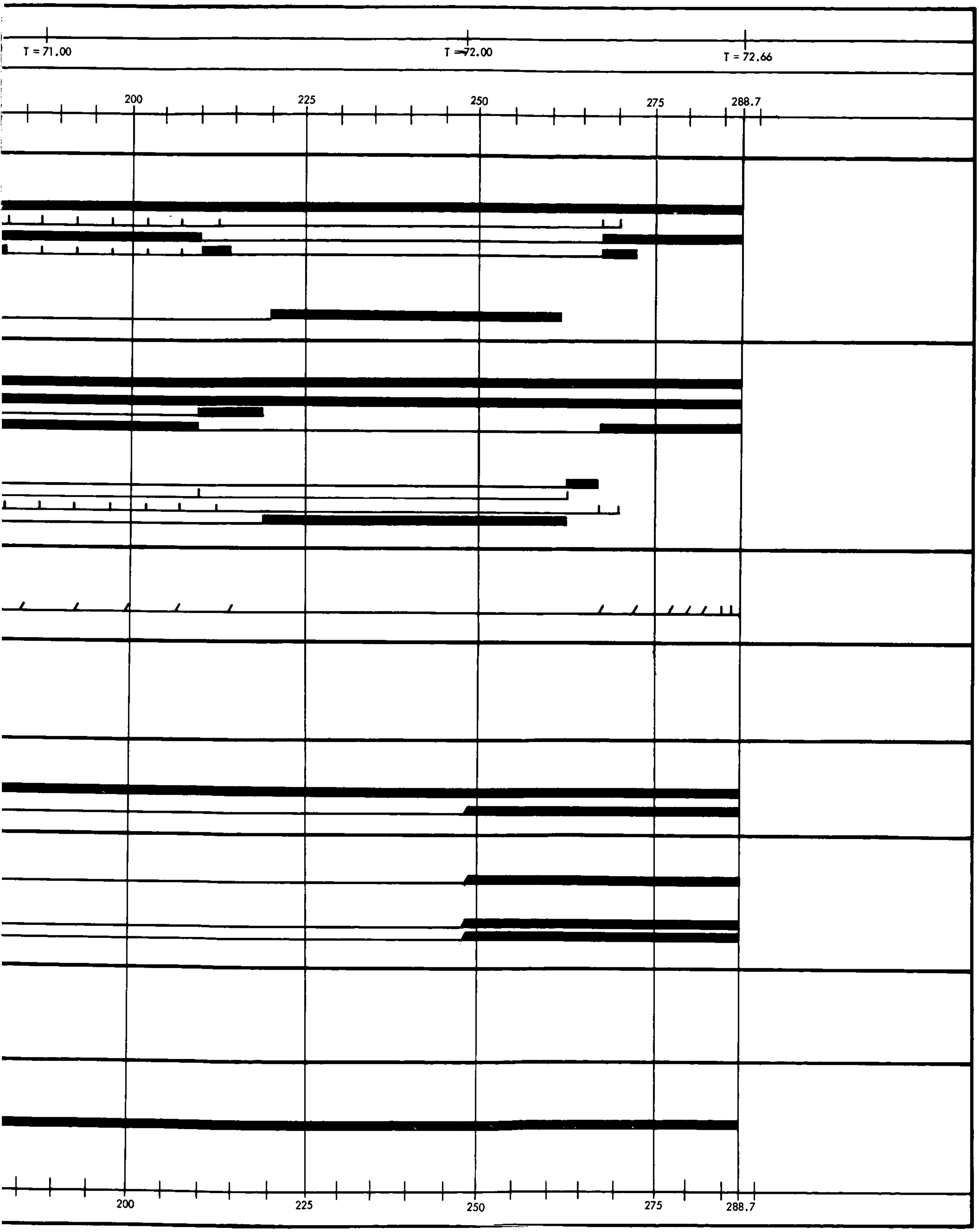


Figure 26. Mission Phase Time Line - Lunar Orbit (Prior to LEM Separation) (Sheet 2 of 2)

[redacted]

[redacted]

LUNAR ORBIT PHASE

(During LEM Landing)

The Lunar Orbit Phase (During LEM Landing) begins with LEM separation and ends with completion of the rendezvous and docking maneuver.

Figure 27 describes the geometry of the LEM injection into an equal period orbit.

Figure 28 describes the geometry of the LEM retro powered descent.

Figure 29 describes the geometry of the LEM final descent to the lunar surface.

Figure 30 is a lunar trace of the C/M and LEM from lunar orbit injection to LEM landing.

Figure 31 describes the geometry of the LEM lunar launch.

Figure 32 describes the geometry of the LEM injection into an elliptical orbit.

Figure 33 describes the geometry of the LEM injection into a circular lunar orbit and rendezvous with the C/M.

Figure 34 is a two-page time-line of spacecraft system activity during this phase.

 [redacted]

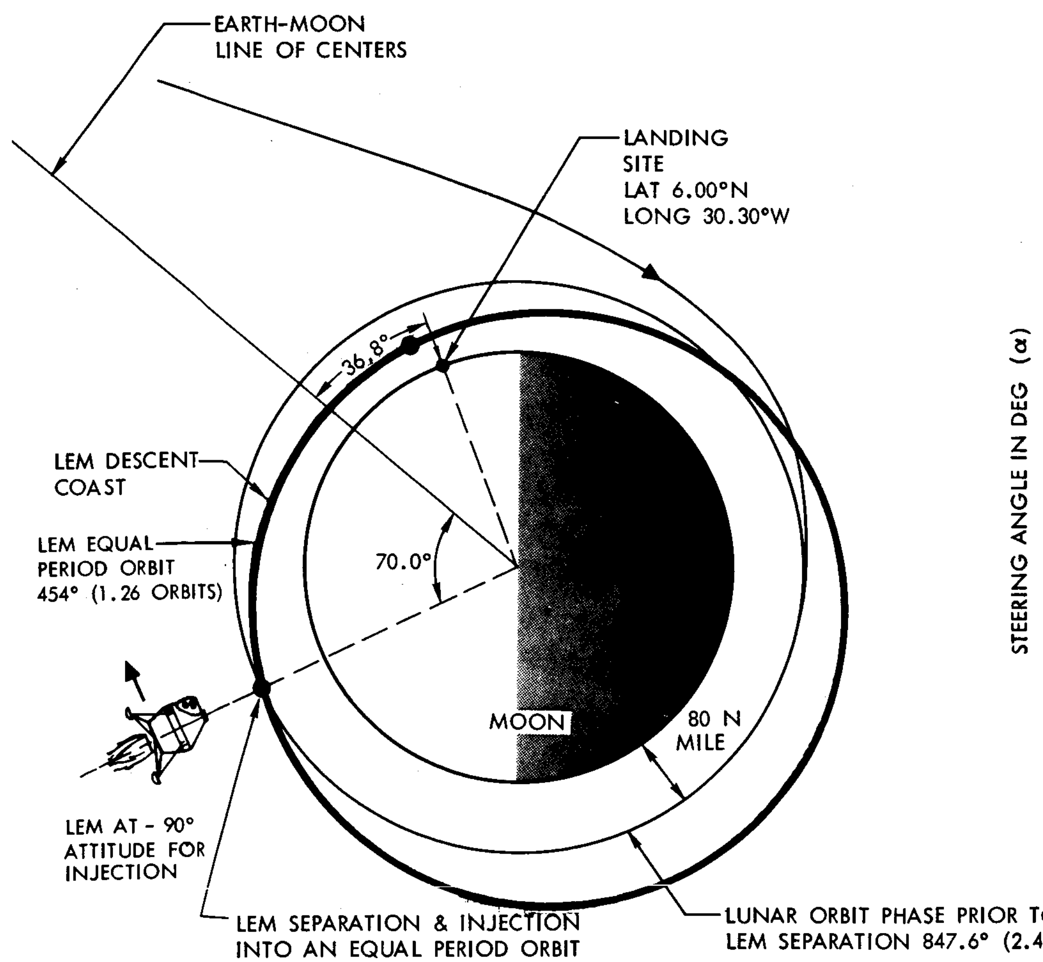
[redacted]

MISSION EVENTS

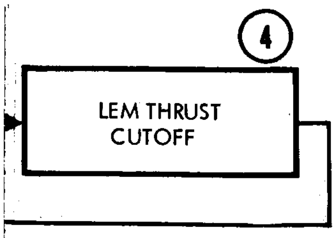
1 LEM SEPARATION → 2 LEM PROPULSION IGNITION → 3 LEM THRUST PERIOD →

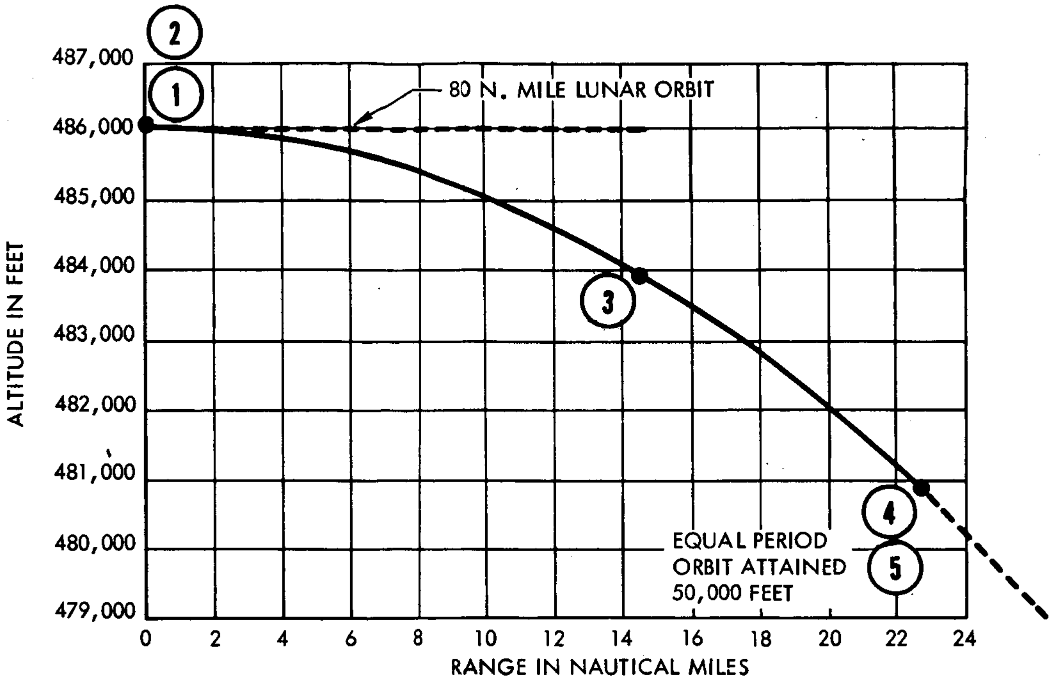
5 BEGIN LEM EQUAL PERIOD ORBIT

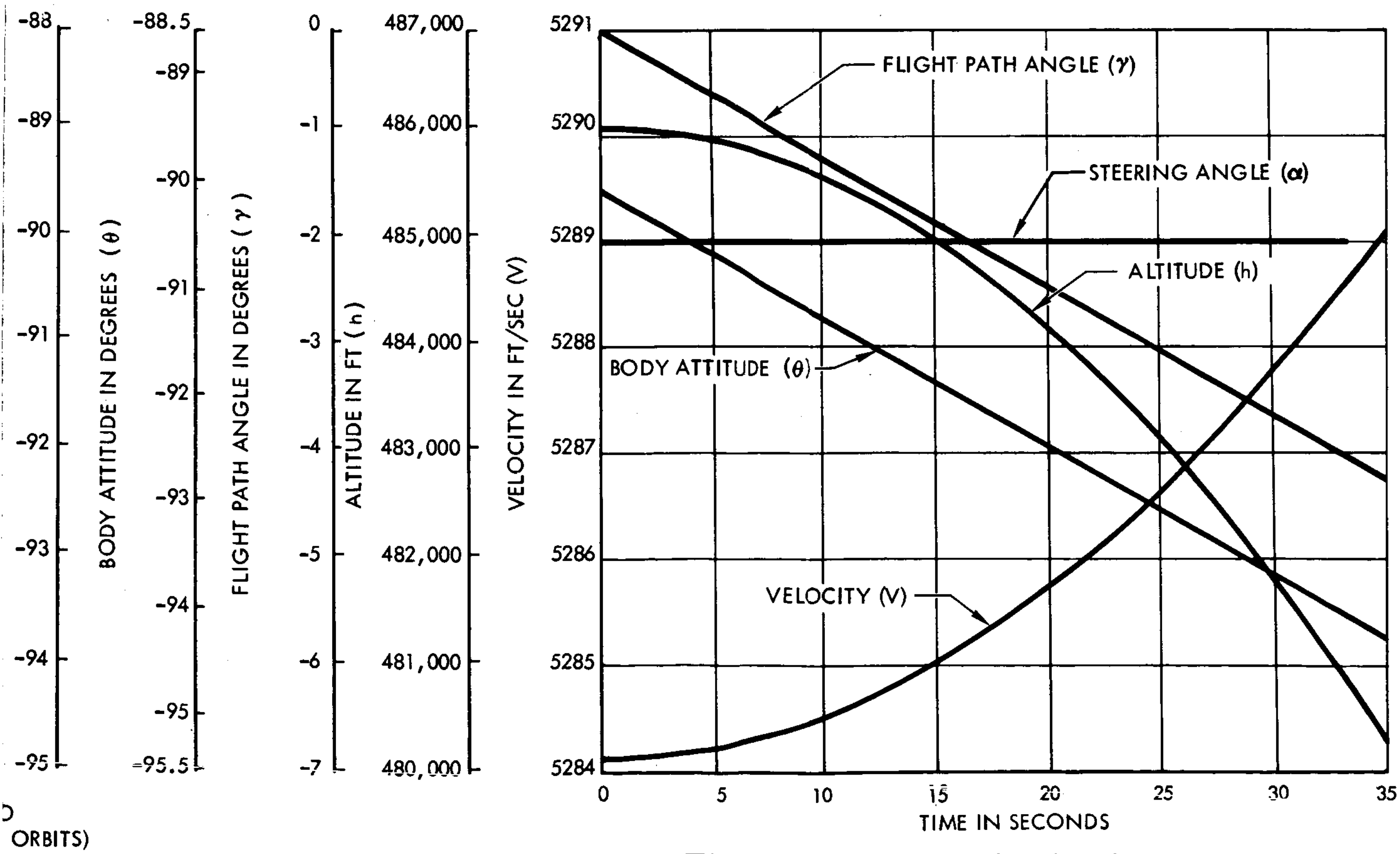
LEM: T/W_0 = .400
ISP = 315 SECONDS
ΔV = 373 FEET PER SECOND



STEERING ANGLE IN DEG (α)







ORBITS)

Figure 27. LEM Injection in Equal Period Orbit

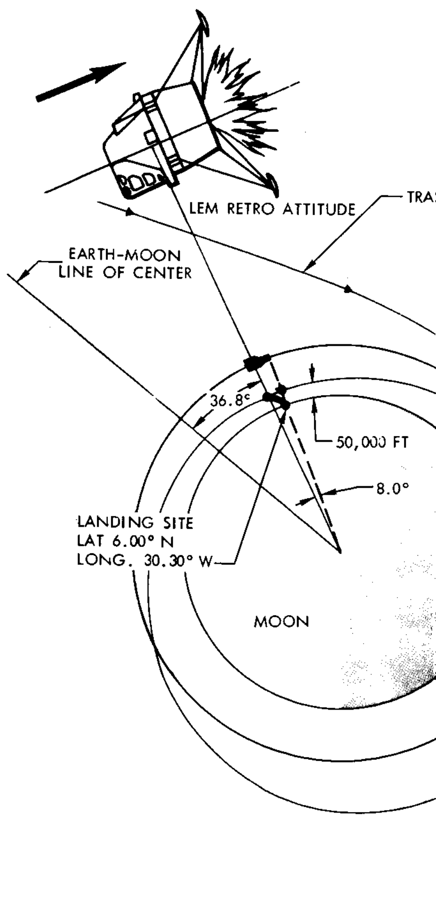
[redacted]

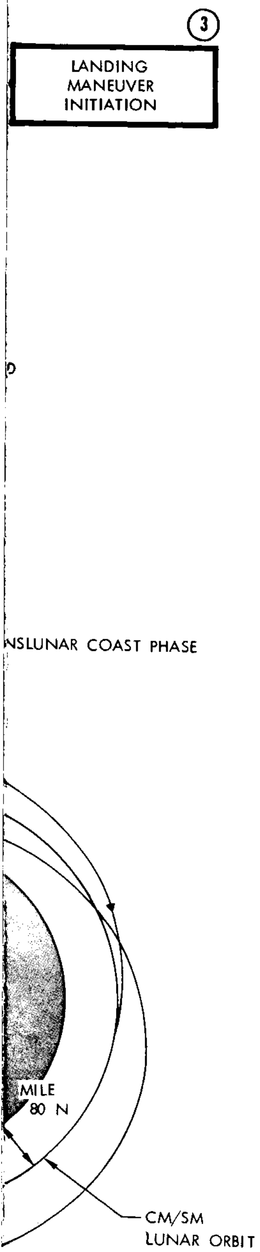
[redacted]

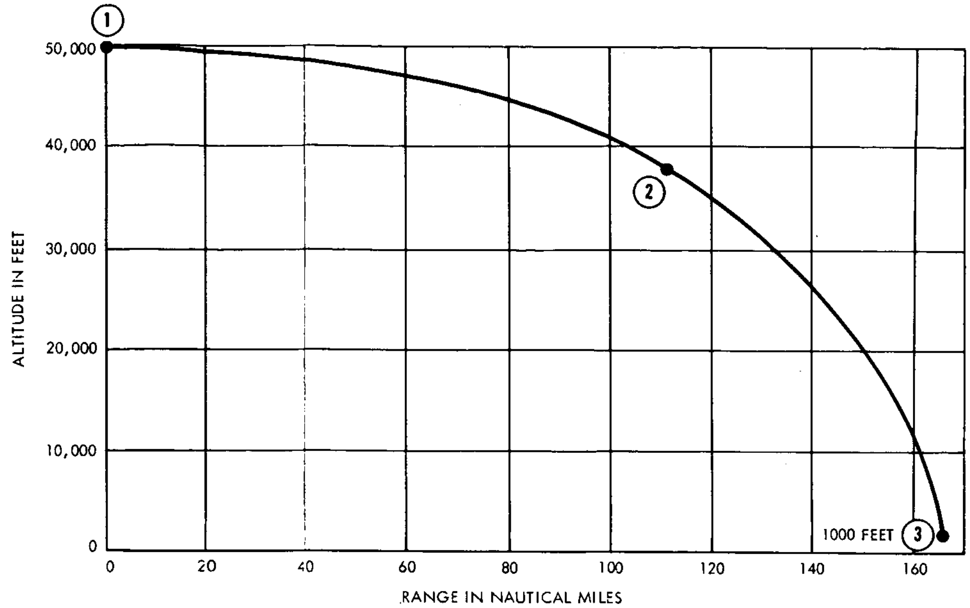
MISSION EVENTS

(1) LEM PROPULSION IGNITION → (2) LEM POWERED DESCENT →

LEM: ISP = 315 SECONDS
T/W_o = .415
ΔV = 5895 FEET PER SECON







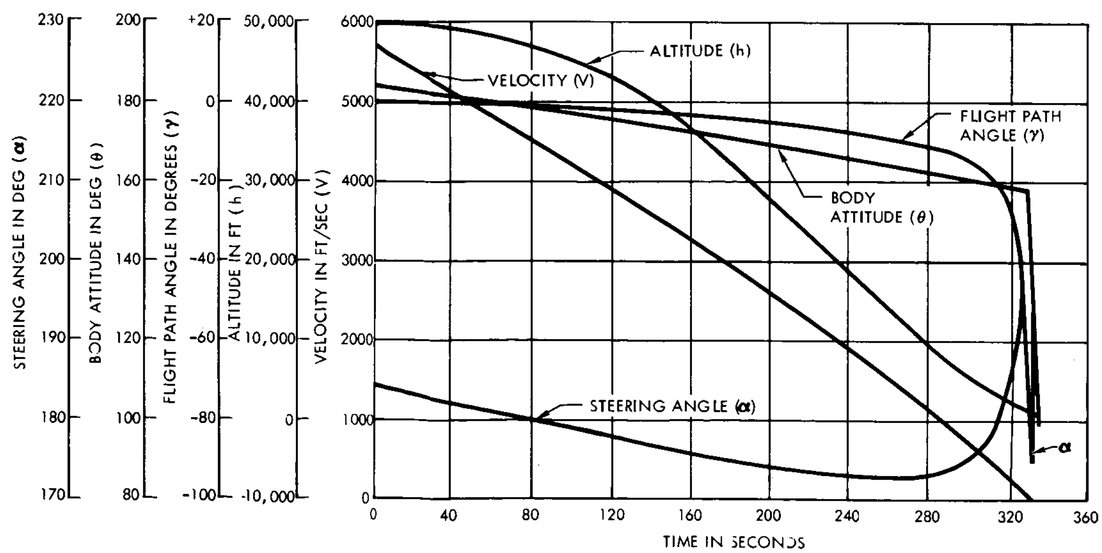


Figure 28. LEM Retro Powered Descent

 [redacted]

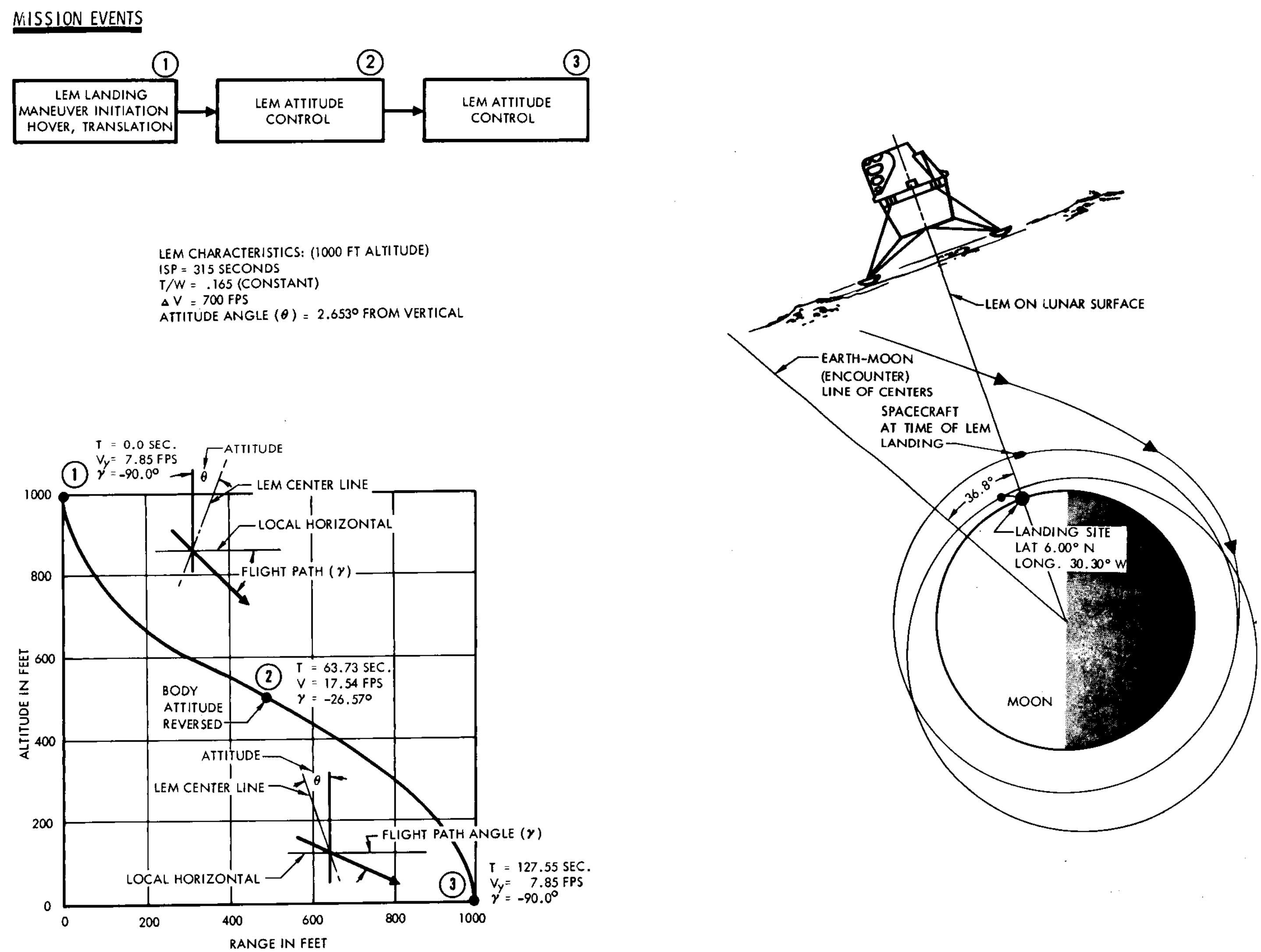
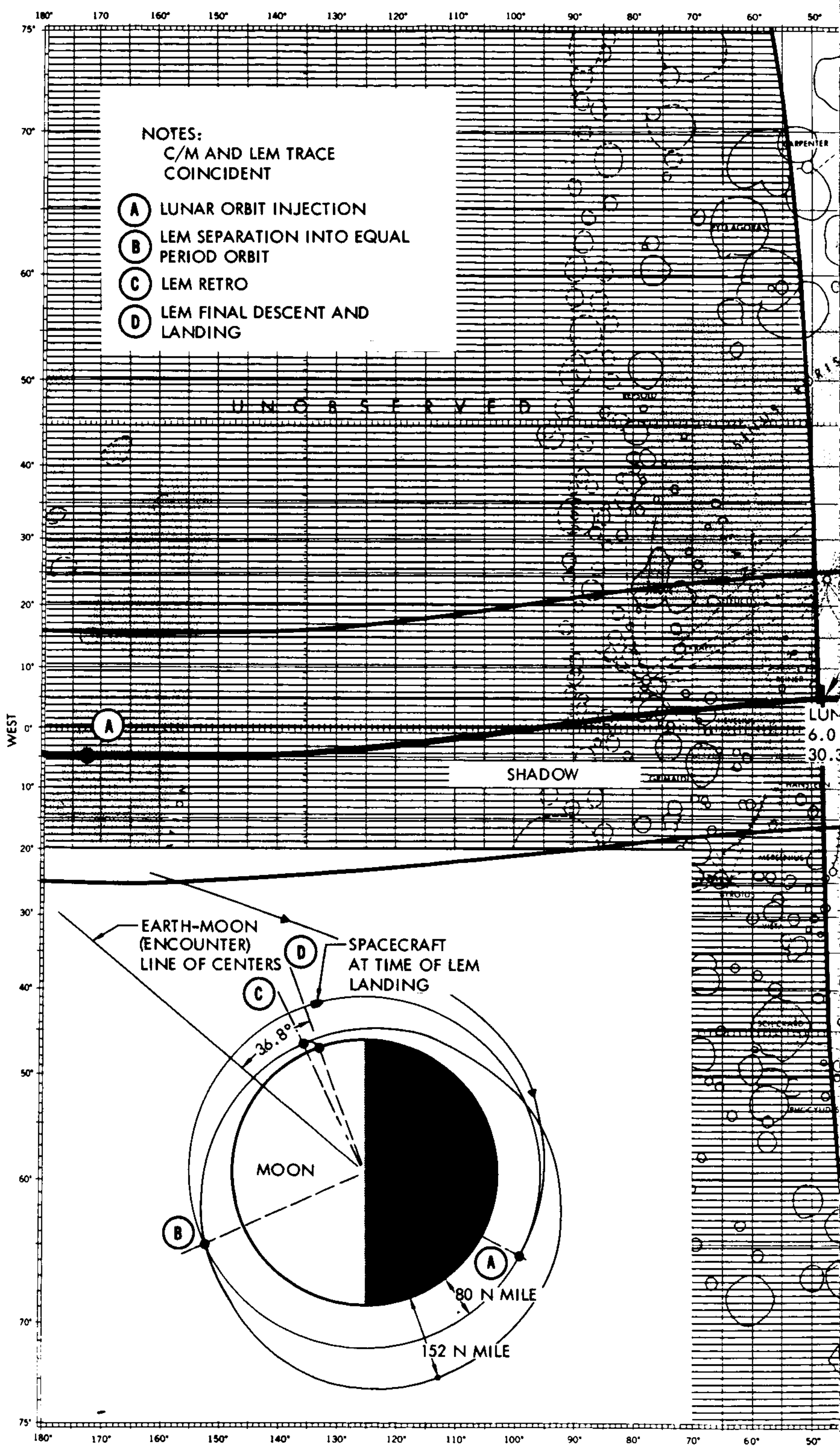


Figure 29. LEM Final Descent

[redacted]

[redacted]



PUBLISHED BY THE AERONAUTICAL CHART AND INFORMATION CENTER, USAF FOR MANNED SPACECRAFT CENTER, NASA
Base Chart Compiled ACIC May 1962

Information on back side of the moon was compiled from the Atlas of The Far Side of The Moon, Published by the Academy of Sciences of the USSR

Figure 30. C/M and L

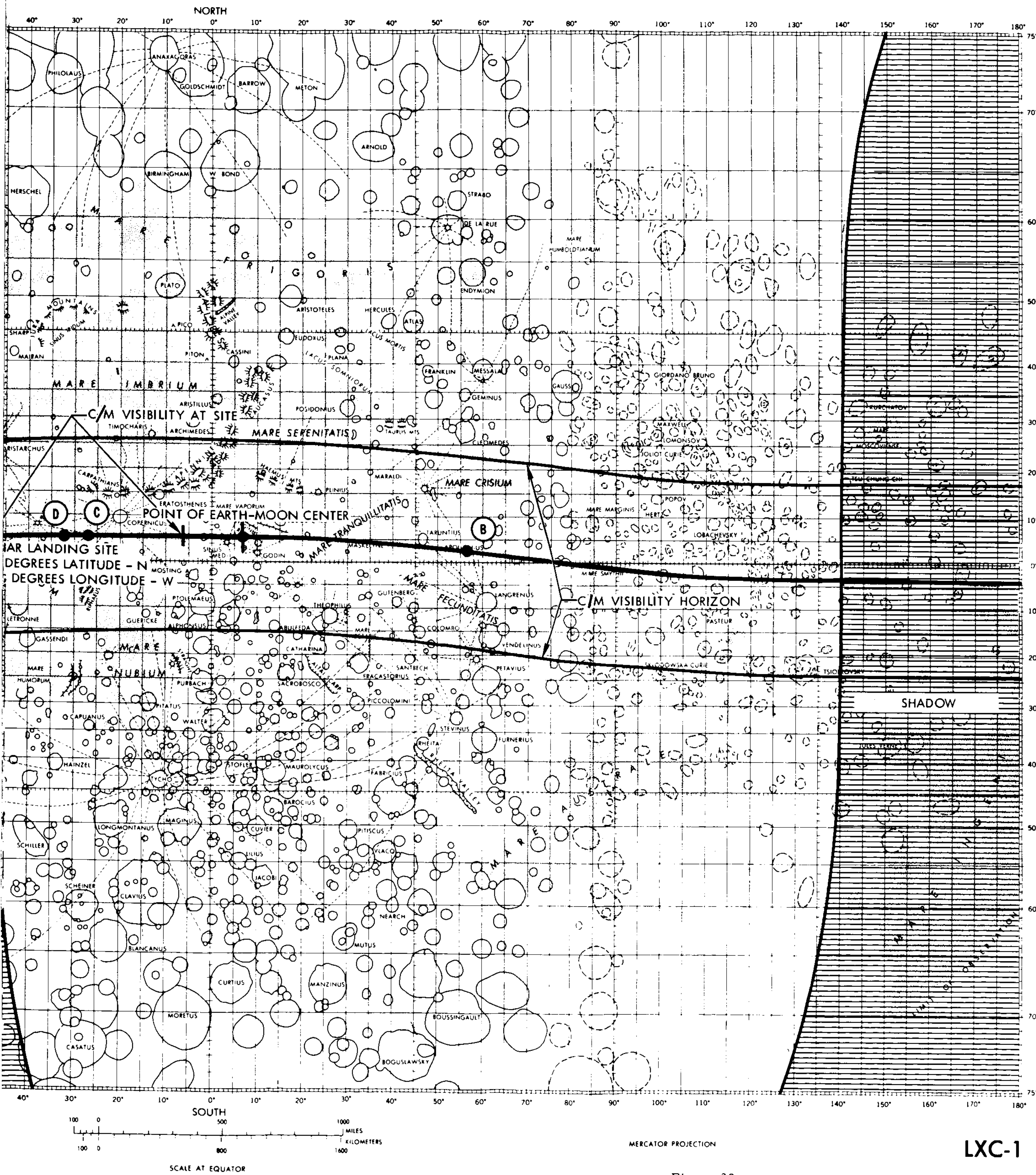


Figure 30.

LEM Lunar Trace (Lunar Orbit Injection To LEM Landing)

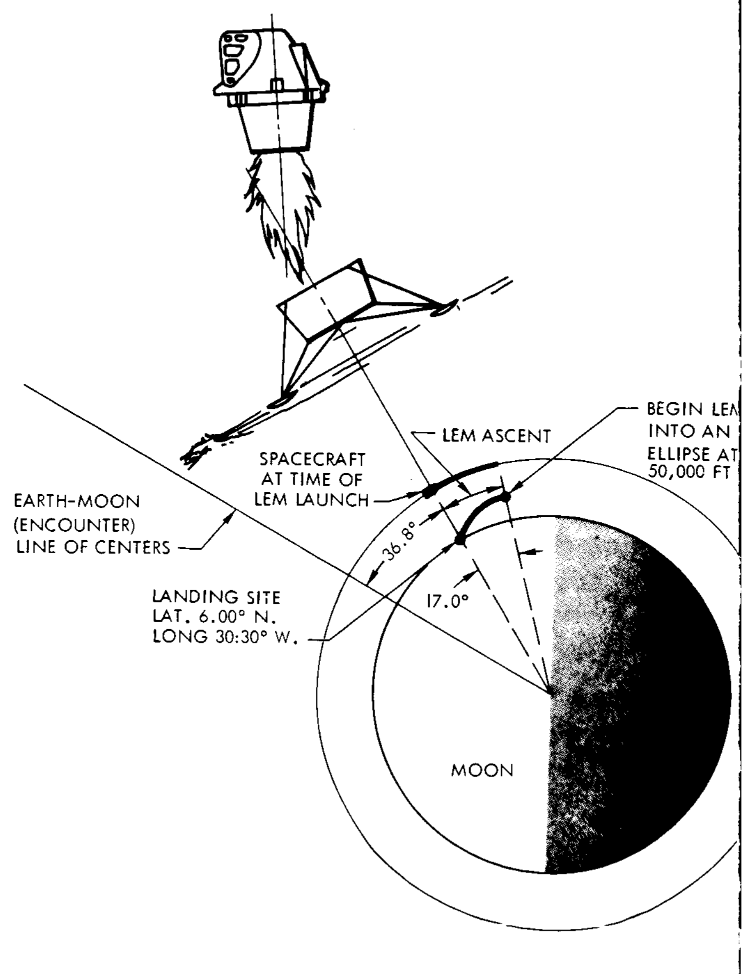
[redacted]

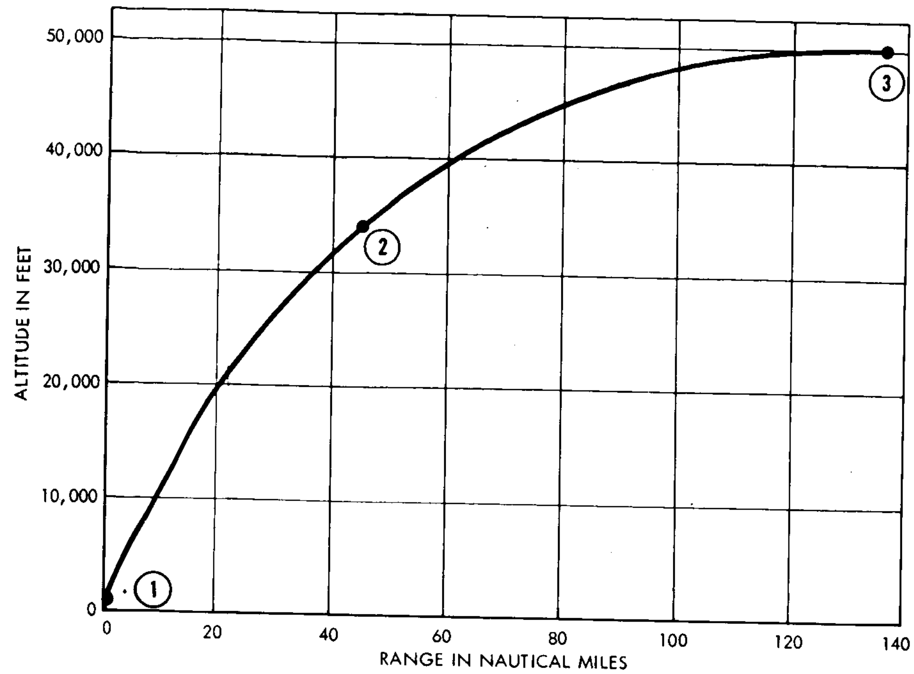
[redacted]

MISSION EVENTS

LEM PROPULSION IGNITION → (1) → LEM ASCENT MANEUVER → (2) → LEM IN INTO E TRANS

LEM: T/W_o = .400
ISP = 315 SECONDS
ΔV = 5806 FEET PER SECOND





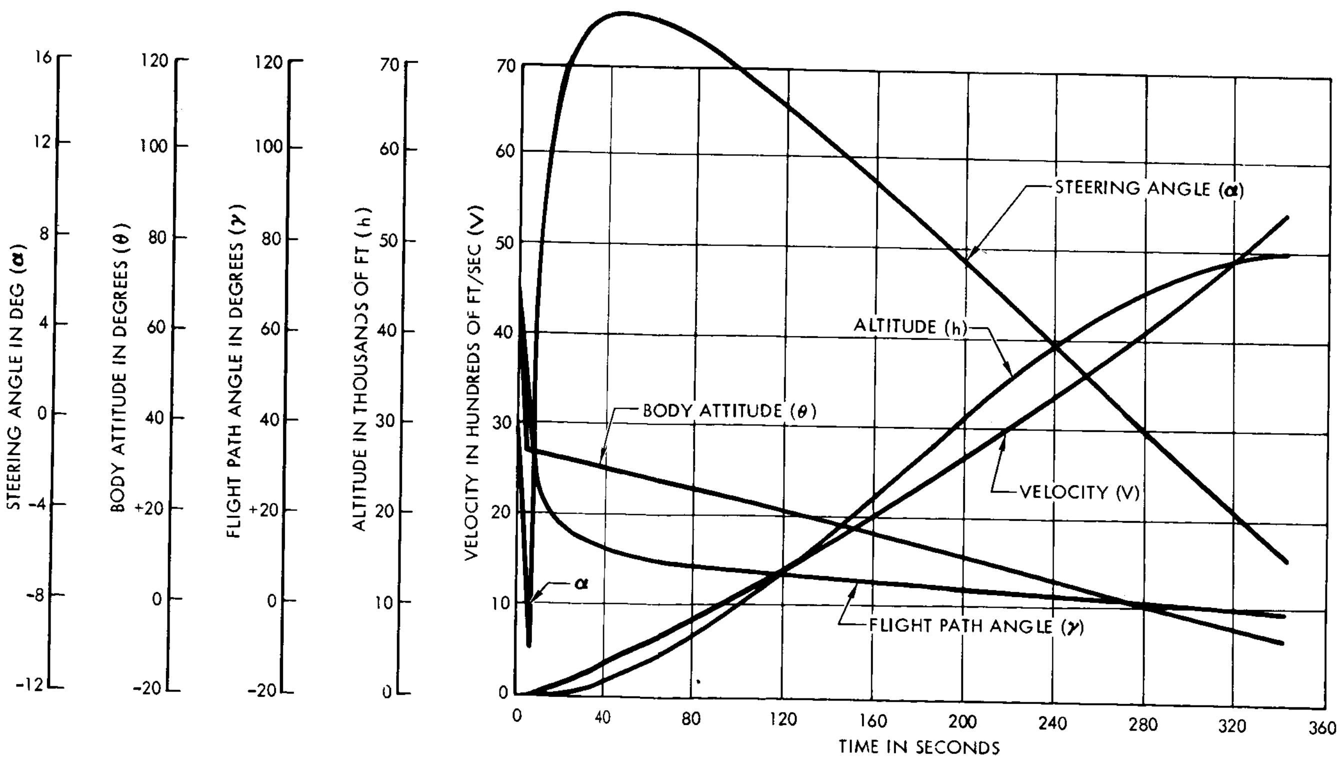


Figure 31. LEM Lunar Launch

~~CONFIDENTIAL~~

[redacted]

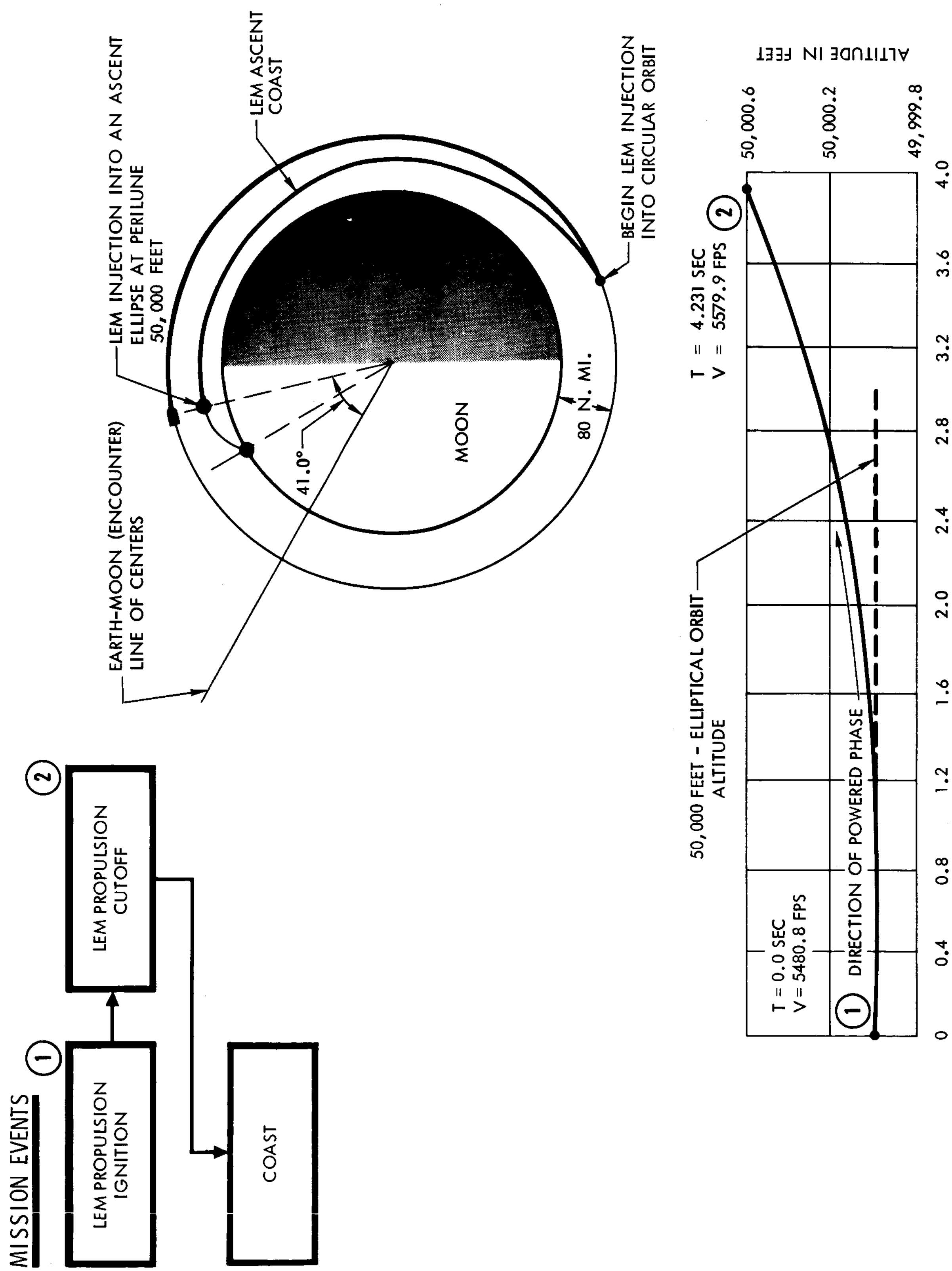


Figure 32. LEM Injection to Ascent Elliptical Orbit

[redacted]

~~CONFIDENTIAL~~

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HRS.	EVENT DURATION
APOLLO S/C SUPPORT OPERATIONS	T = 72.66	609 MIN
LEM SEPARATION	T = 72.66	
RECORDING & RELAYING LEM PROGRESS		
PHOTOGRAPHING LEM PROGRESS		
LUNAR SURVEILLANCE		
NAVIGATIONAL DATA PROCESSING FOR RENDEZVOUS		
VERIFYING RENDEZVOUS PREPARATIONS		
RENDEZVOUS MONITORING		
DOCKING STABILIZATION		
LEM LUNAR LANDING & RENDEZVOUS	T = 72.66	
SEPARATION FROM S/C	T = 72.66	
ATTITUDE MANEUVER		
ELIPTICAL ORBIT INJECTION (36 SEC) & COAST		154 MIN
POST INJECTION CHECK		5 MIN
ELIPTICAL ORBIT PARAMETERS CHECK	T = 72.77	50 MIN
LUNAR LANDING COUNTDOWN	T = 74.99	15 MIN
RETROGRADE MANEUVER	T = 75.25	336 SEC
HOVER/TRANSLATION	T = 75.34	120 SEC
TOUCHDOWN	T = 75.37	
POST LANDING CHECKOUT & ARRANGEMENT		1 HR
EARTH COMMUNICATION		1 HR
EXPLORATION & PHENOMENOM MEASUREMENTS		215 MIN
RENDEZVOUS PARAMETERS CHECK		1 HR
LUNAR LAUNCH PREPARATIONS	T = 77.66	30 MIN
LEM COUNTDOWN		15 MIN
ASCENT MANEUVER	T = 81.24	356 SEC
ELIPTICAL ORBIT COAST	T = 81.34	58 MIN
PARKING ORBIT INJECTION, RENDEZVOUS, & DOCKING	T = 82.31	30 MIN
LEM DOCKED & LATCHED	T = 82.81	

GOSS DSIF COVERAGE - ESTIMATED

GOLDSTONE
JOHANNESBURG
WOOMERA

LEM - DSIF COVERAGE - ESTIMATED

GOLDSTONE
JOHANNESBURG
WOOMERA

POSITIONAL DATA - ESTIMATED

S/C OVER LUNAR NIGHT

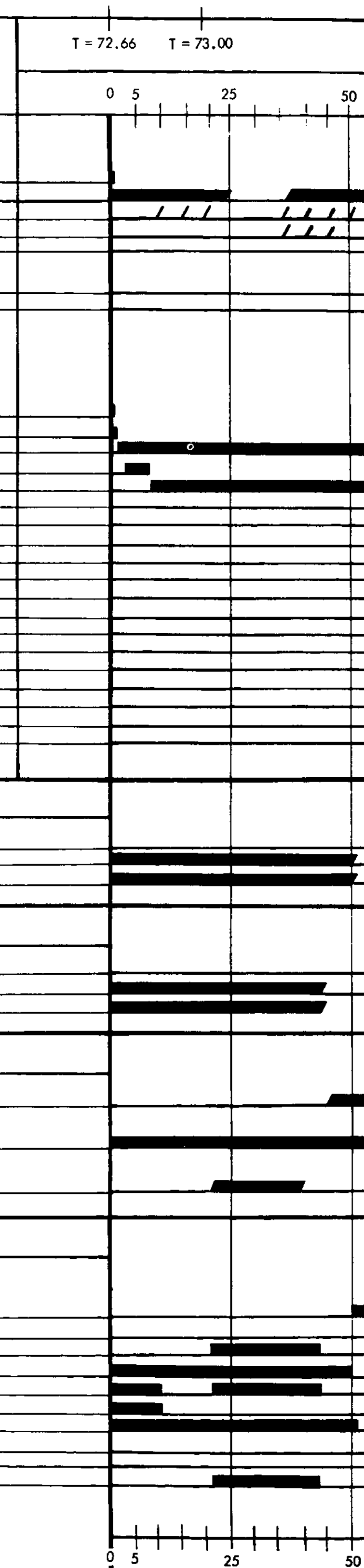
S/C - LEM COMMUNICATION

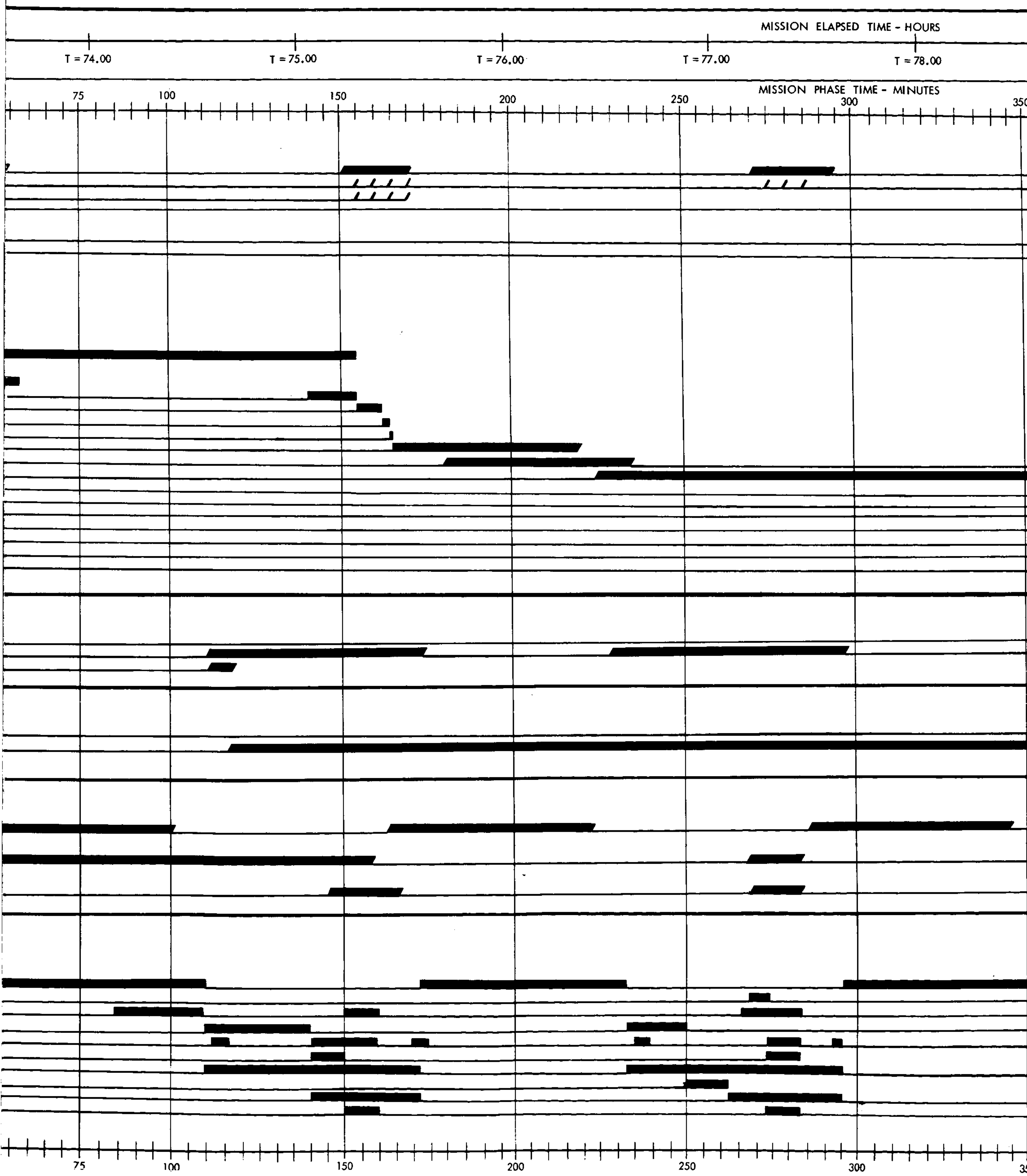
S/C IN LINE OF SIGHT WITH LANDING SITE

PERTINENT FUNCTIONS

COMMUNICATIONS & INSTRUMENTATION SYSTEM

DATA STORAGE RECORDING
2-WAY VOICE WITH BELT PACKS
2-WAY VOICE WITH LEM
DSIF NARROW BAND TELEMETRY
DSIF 2-WAY VOICE WITH GOSS
C/M DSIF TV TRANSMISSION
DSIF 2-WAY DOPPLER TRACKING/RANGING
DSIF BROAD BAND TELEMETRY
DSIF DATA STORAGE TRANSMISSION
DSIF 2-WAY VOICE RELAY TO GOSS




MISSION ELAPSED TIME - HOURS
T = 74.00
T = 75.00
T = 76.00
T = 77.00
T = 78.00
MISSION PHASE TIME - MINUTES
75
100
150
200
250
300
350

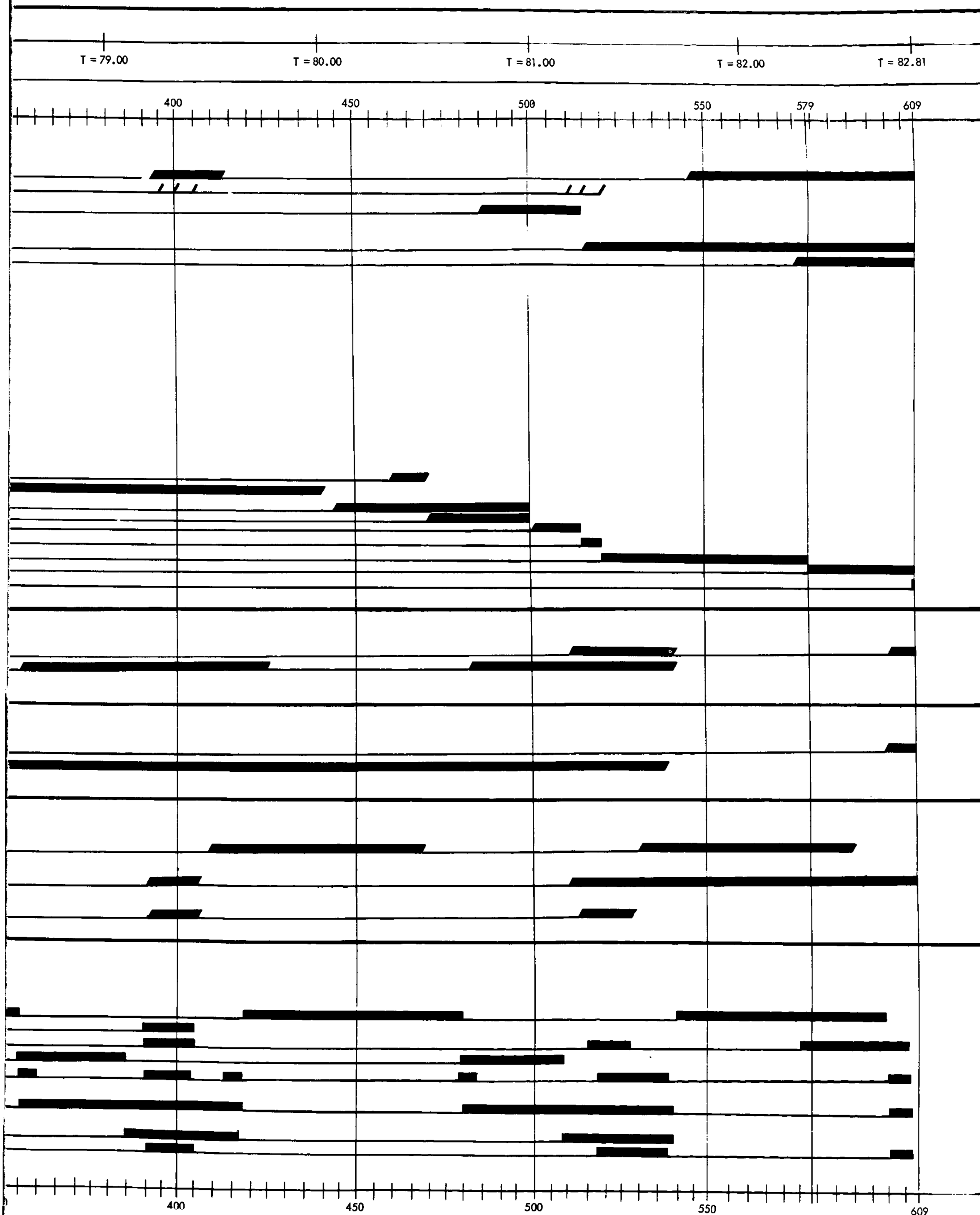
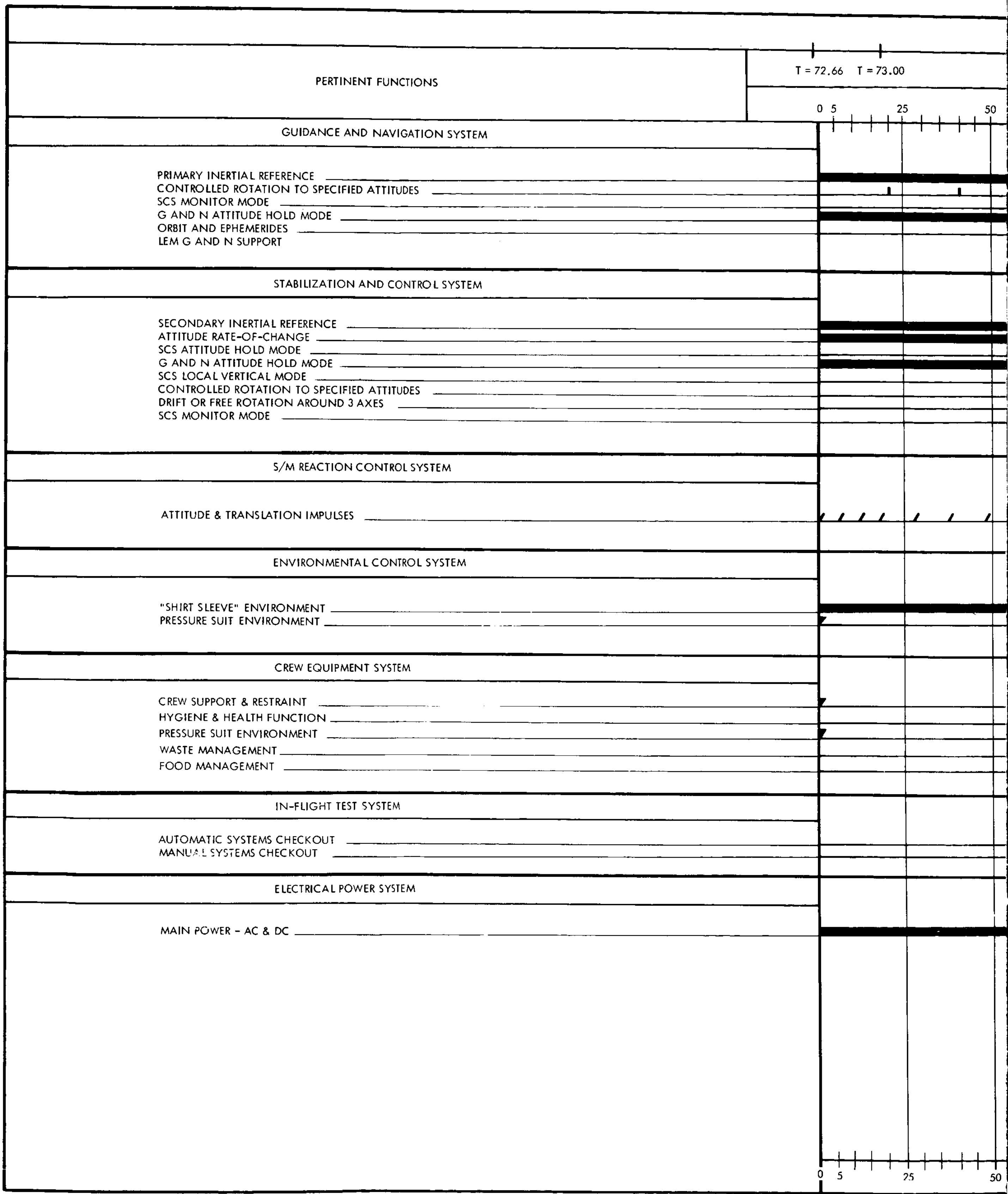


Figure 34. Mission Phase Time Line-Lunar Orbit (During LEM Landing) (Sheet 1 of 2)

[redacted]


PERTINENT FUNCTIONS
T = 72.66
T = 73.00
0
5
25
50
GUIDANCE AND NAVIGATION SYSTEM
PRIMARY INERTIAL REFERENCE
CONTROLLED ROTATION TO SPECIFIED ATTITUDES
SCS MONITOR MODE
G AND N ATTITUDE HOLD MODE
ORBIT AND EPHEMERIDES
LEM G AND N SUPPORT
STABILIZATION AND CONTROL SYSTEM
SECONDARY INERTIAL REFERENCE
ATTITUDE RATE-OF-CHANGE
SCS ATTITUDE HOLD MODE
G AND N ATTITUDE HOLD MODE
SCS LOCAL VERTICAL MODE
CONTROLLED ROTATION TO SPECIFIED ATTITUDES
DRIFT OR FREE ROTATION AROUND 3 AXES
SCS MONITOR MODE
S/M REACTION CONTROL SYSTEM
ATTITUDE & TRANSLATION IMPULSES
ENVIRONMENTAL CONTROL SYSTEM
"SHIRT SLEEVE" ENVIRONMENT
PRESSURE SUIT ENVIRONMENT
CREW EQUIPMENT SYSTEM
CREW SUPPORT & RESTRAINT
HYGIENE & HEALTH FUNCTION
PRESSURE SUIT ENVIRONMENT
WASTE MANAGEMENT
FOOD MANAGEMENT
IN-FLIGHT TEST SYSTEM
AUTOMATIC SYSTEMS CHECKOUT
MANUAL SYSTEMS CHECKOUT
ELECTRICAL POWER SYSTEM
MAIN POWER - AC & DC
0
5
25
50


MISSION ELAPSED TIME - HOURS
T = 74.00
T = 75.00
T = 76.00
T = 77.00
T = 78.00
MISSION PHASE TIME - MINUTES
75
100
125
150
175
200
225
250
275
300
325
350
75
100
200
300

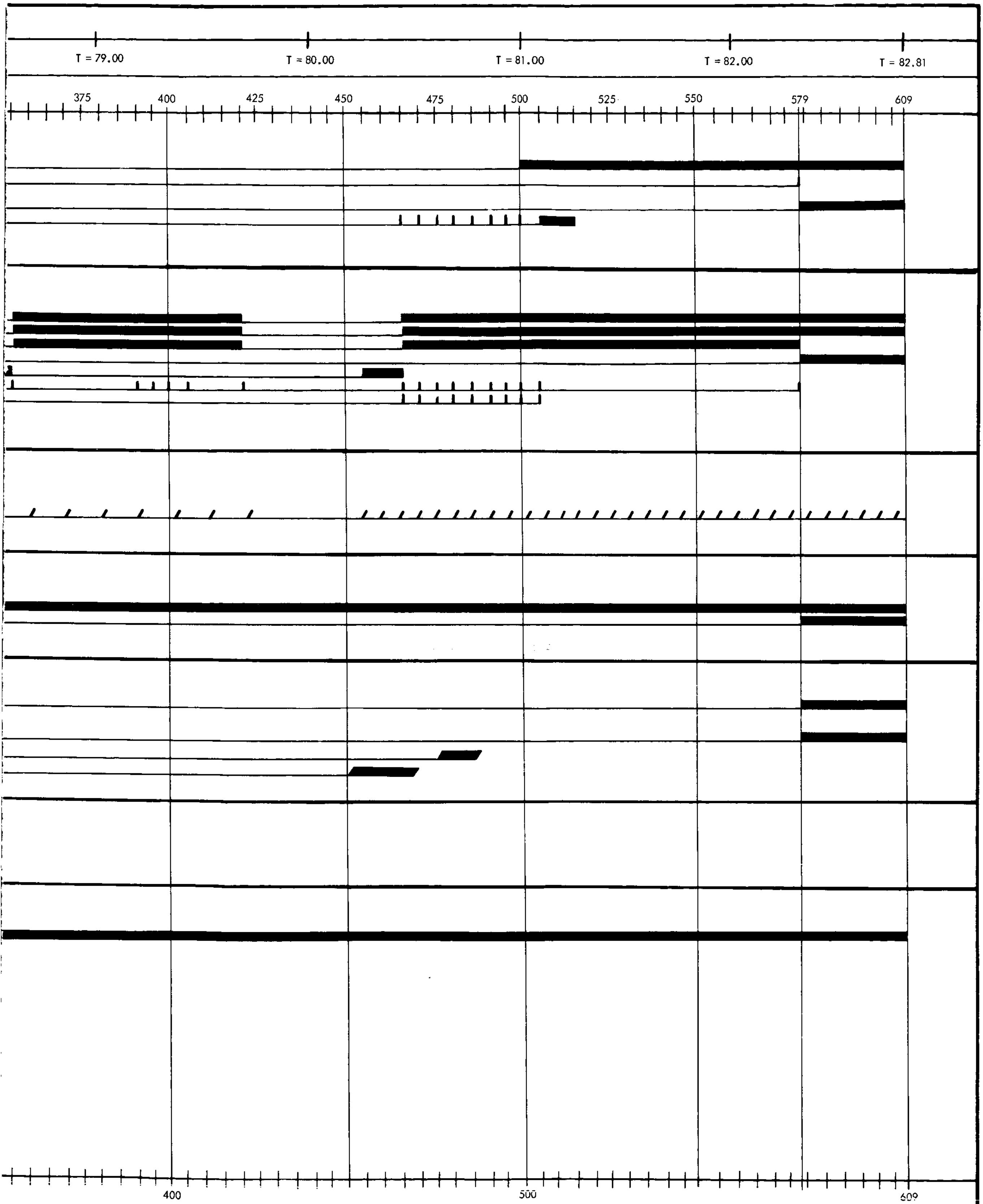


Figure 34. Mission Phase Time Line-Lunar Orbit (During LEM Landing) (Sheet 2 of 2)

LUNAR ORBIT PHASE

(Subsequent to LEM Rendezvous)

The Lunar Orbit Phase (Subsequent to LEM Rendezvous) begins with Post Docking Check and ends with S/M Reaction Control System ullage acceleration.

Figure 35 describes the geometry of this phase.

Figure 36 is a lunar trace of the C/M and LEM from lunar launch to transearth injection.

Figure 37 is a two-page time-line delineation of spacecraft system activity during this phase.

[redacted]

MISSION EVENTS

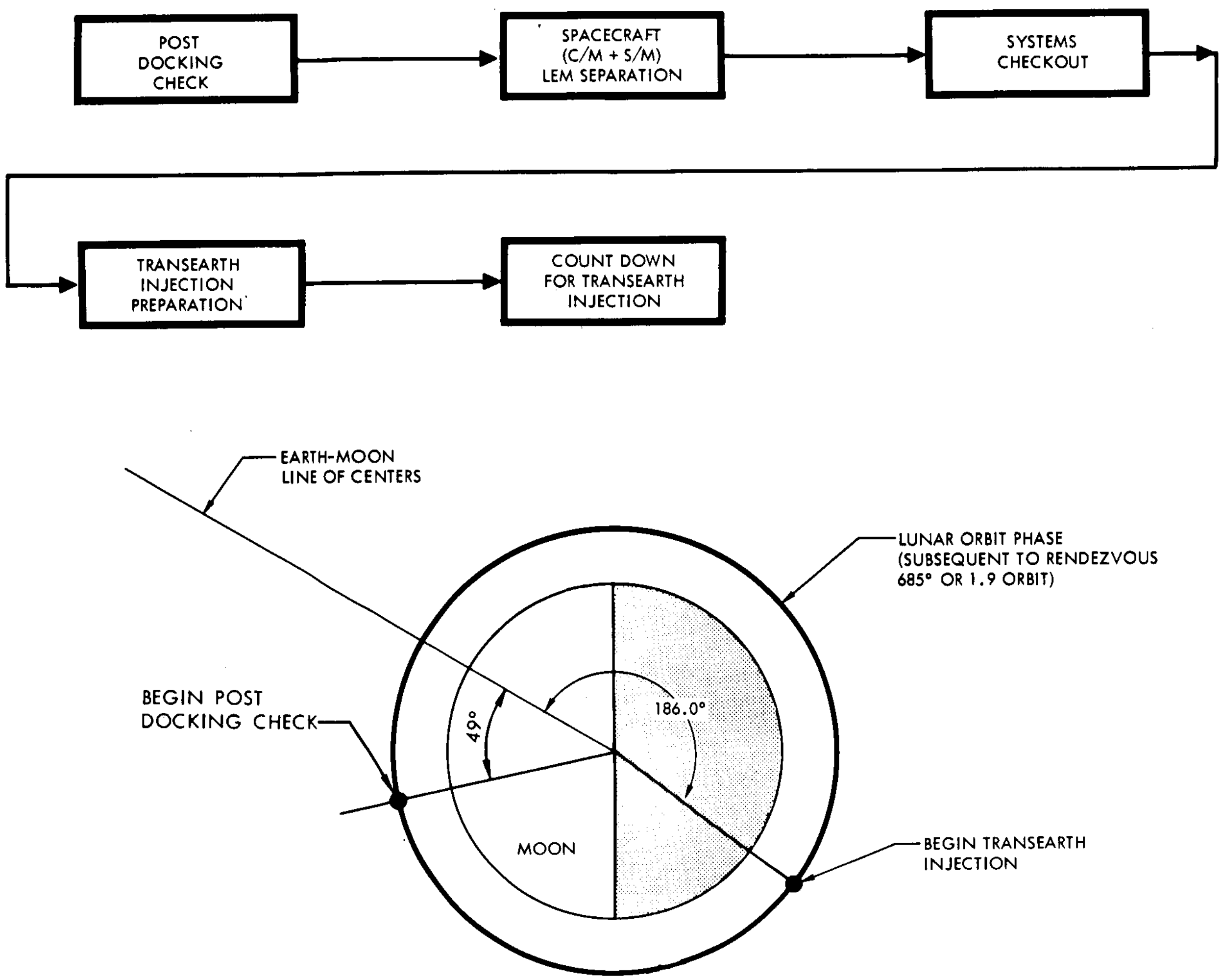
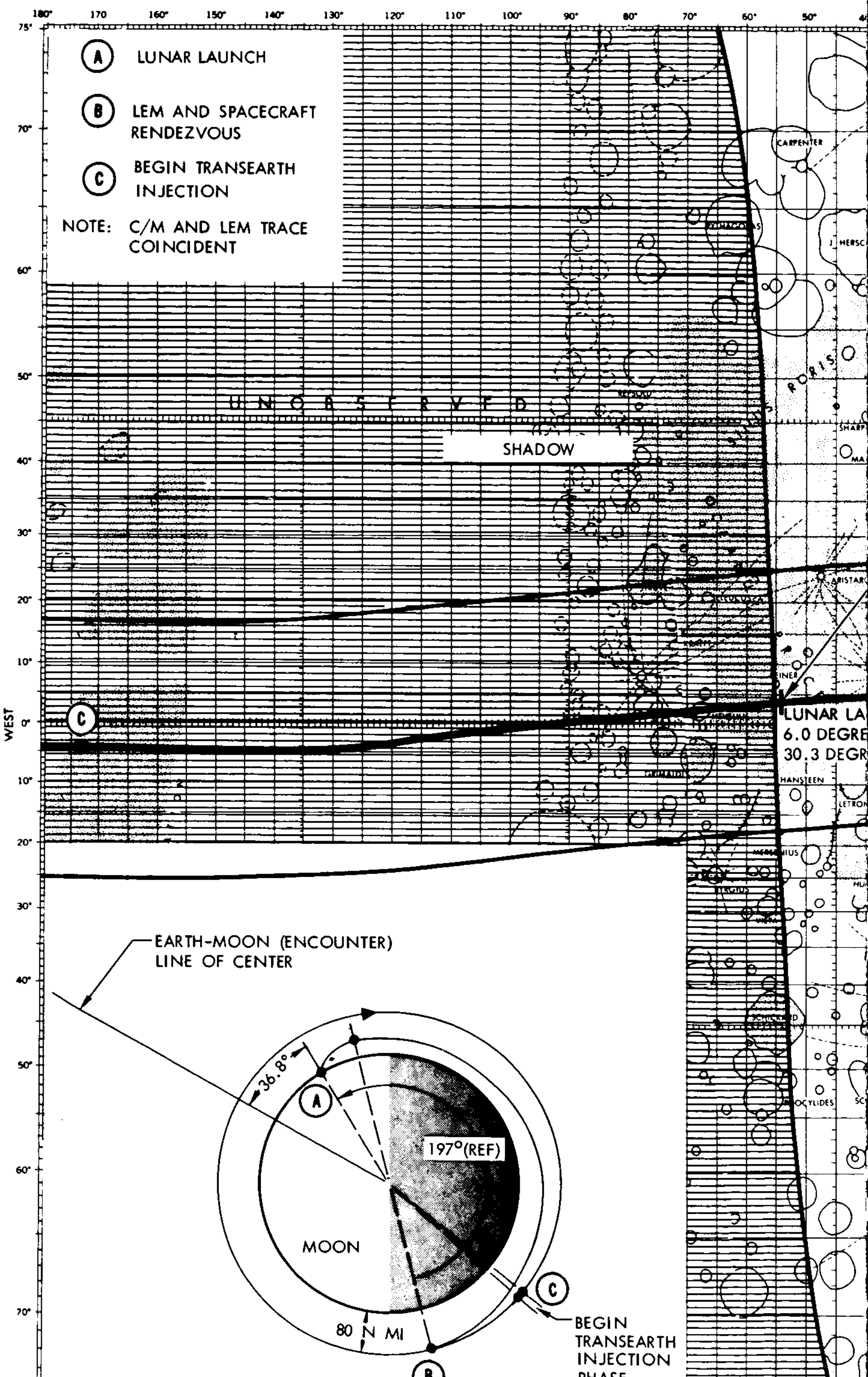


Figure 35. Lunar Orbit Phase (Subsquent to LEM Rendezvous)

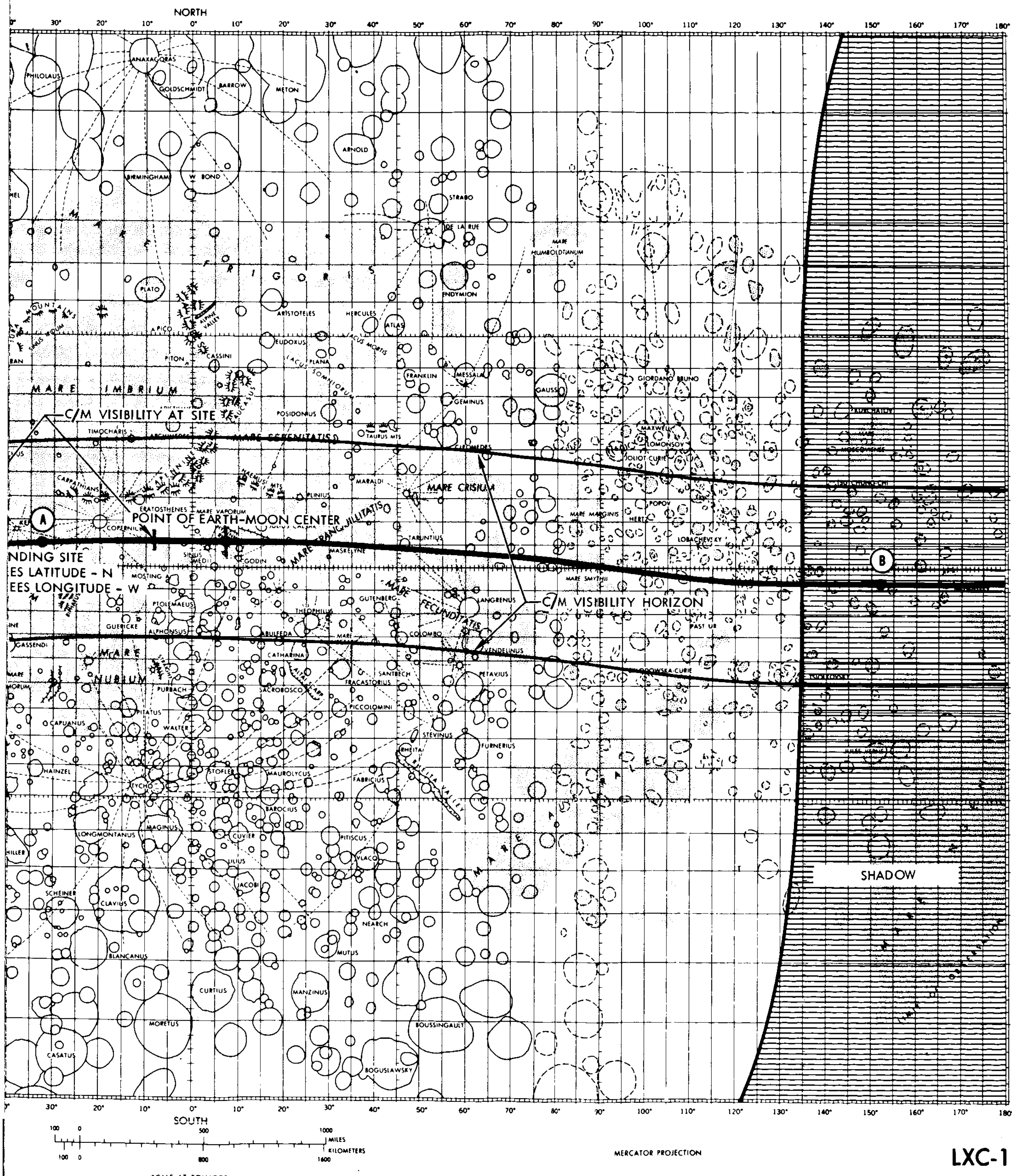
[redacted]



PUBLISHED BY THE AERONAUTICAL CHART AND INFORMATION CENTER, USAF FOR MANNED SPACECRAFT CENTER, NASA
Base Chart Compiled ACIC May 1962

Information on back side of the moon was compiled from the Atlas of The Far Side of The Moon, Published by the Academy of Sciences of the USSR

Figure 36. C/M and LEM Tra



LXC-1

...jectory Lunar Trace (Lunar Launch To Transearth Injection)

~~CONFIDENTIAL~~

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION
POST DOCKING CHECK AND ARRANGEMENT	T = 82.81	25 MIN
SECURE LEM SYSTEMS		
CREW & EQUIPMENT TRANSFER TO C/M		
C/M - LEM SEPARATION		5 MIN
SYSTEMS CHECKOUT	T = 83.27	20 MIN
TRANSEARTH INJECTION PREPARATION	T = 83.31	160 MIN
LUNAR ORBIT PARAMETERS CHECK		
TRANSEARTH INJECTION PARAMETERS COMPUTATION		
IMU FINE ALIGNMENT		
COUNTDOWN FOR TRANSEARTH INJECTION	T = 85.97	15 MIN
GIMBAL ACTIVATION & PRESETTING		
PROPULSION SYSTEM ARMING		
OTHER SYSTEMS SET UP		
EQUIPMENT & CREW ARRANGEMENT		
S/C ORIENTED FOR INJECTION		
TRANSEARTH INJECTION START SIGNAL	T = 86.20	

GOSS DSIF COVERAGE - ESTIMATED

GOLDSTONE

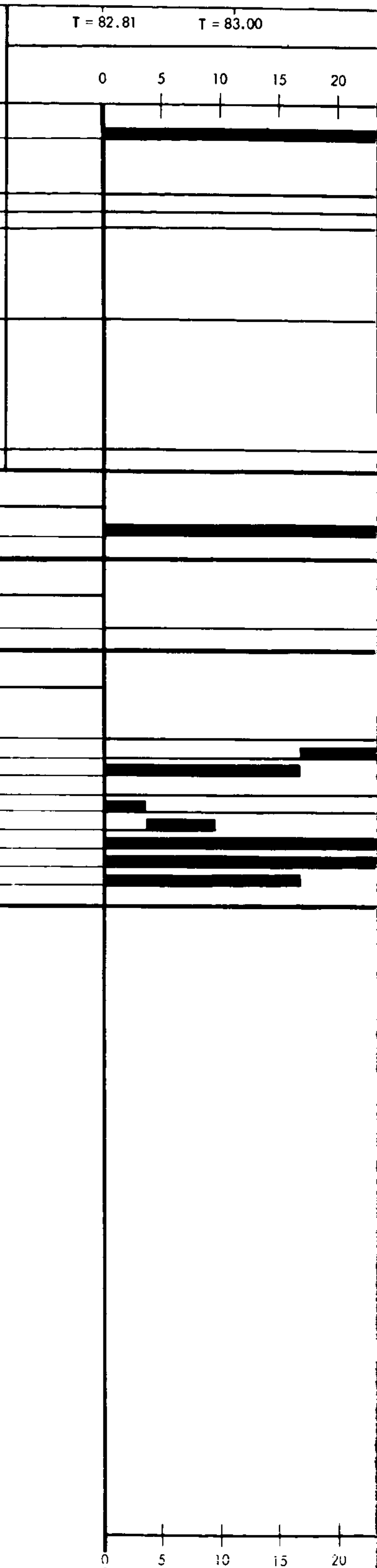
POSITIONAL DATA - ESTIMATED

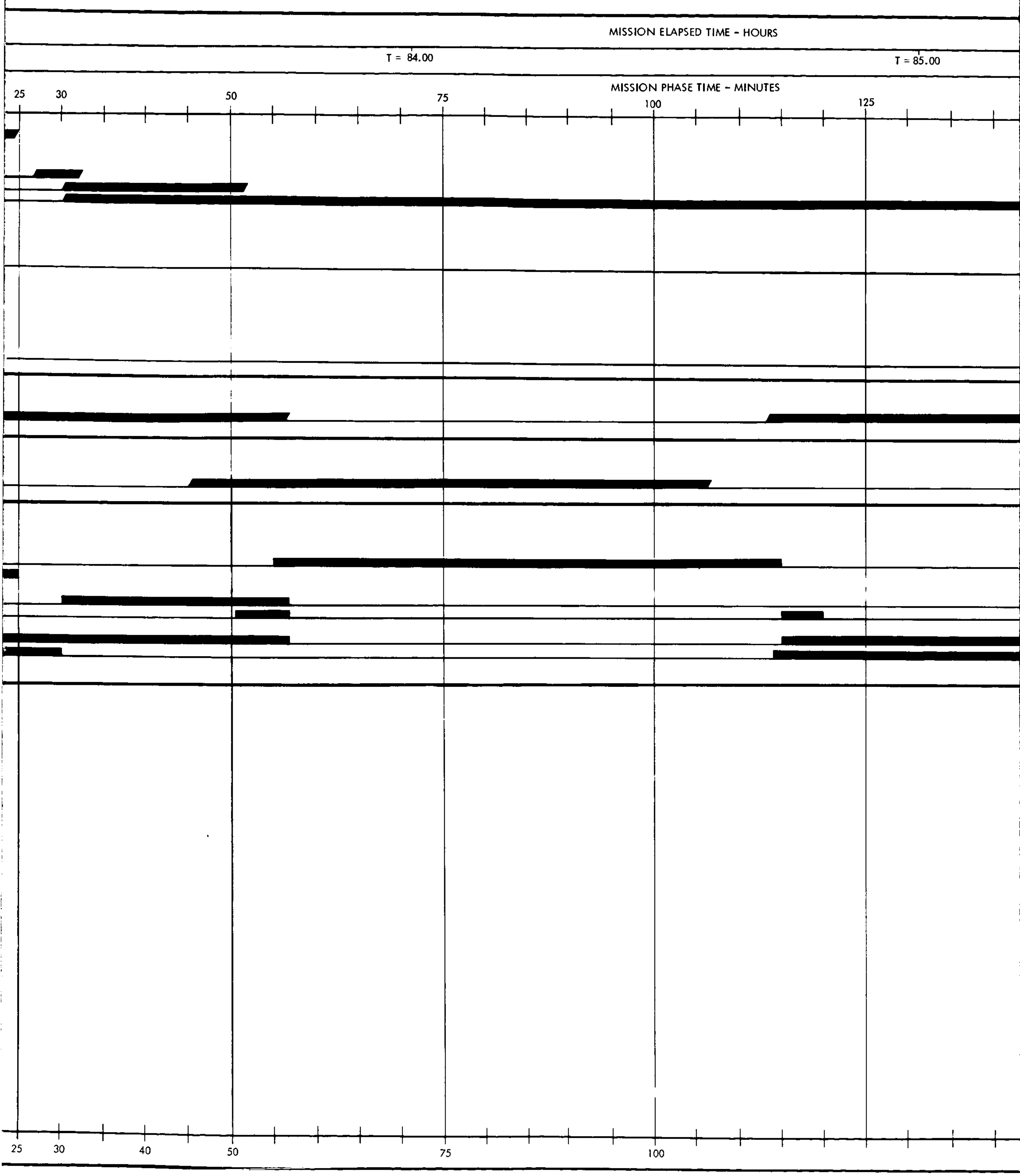
OVER LUNAR NIGHT

PERTINENT FUNCTIONS

COMMUNICATIONS & INSTRUMENTATION SYSTEM

- DATA STORAGE RECORDING
- TWO WAY VOICE WITH BELT PACKS
- TWO WAY VOICE WITH LEM
- DSIF NARROW BAND TELEMETRY
- DSIF 2-WAY VOICE WITH GOSS
- C/M DSIF TV TRANSMISSION
- DSIF 2-WAY DOPPLER TRACKING/RANGING
- DSIF DATA STORAGE TRANSMISSION
- DSIF 2-WAY VOICE RELAY TO GOSS




MISSION ELAPSED TIME - HOURS
T = 84.00
T = 85.00
MISSION PHASE TIME - MINUTES
25
30
50
75
100
125
25
30
40
50
75
100

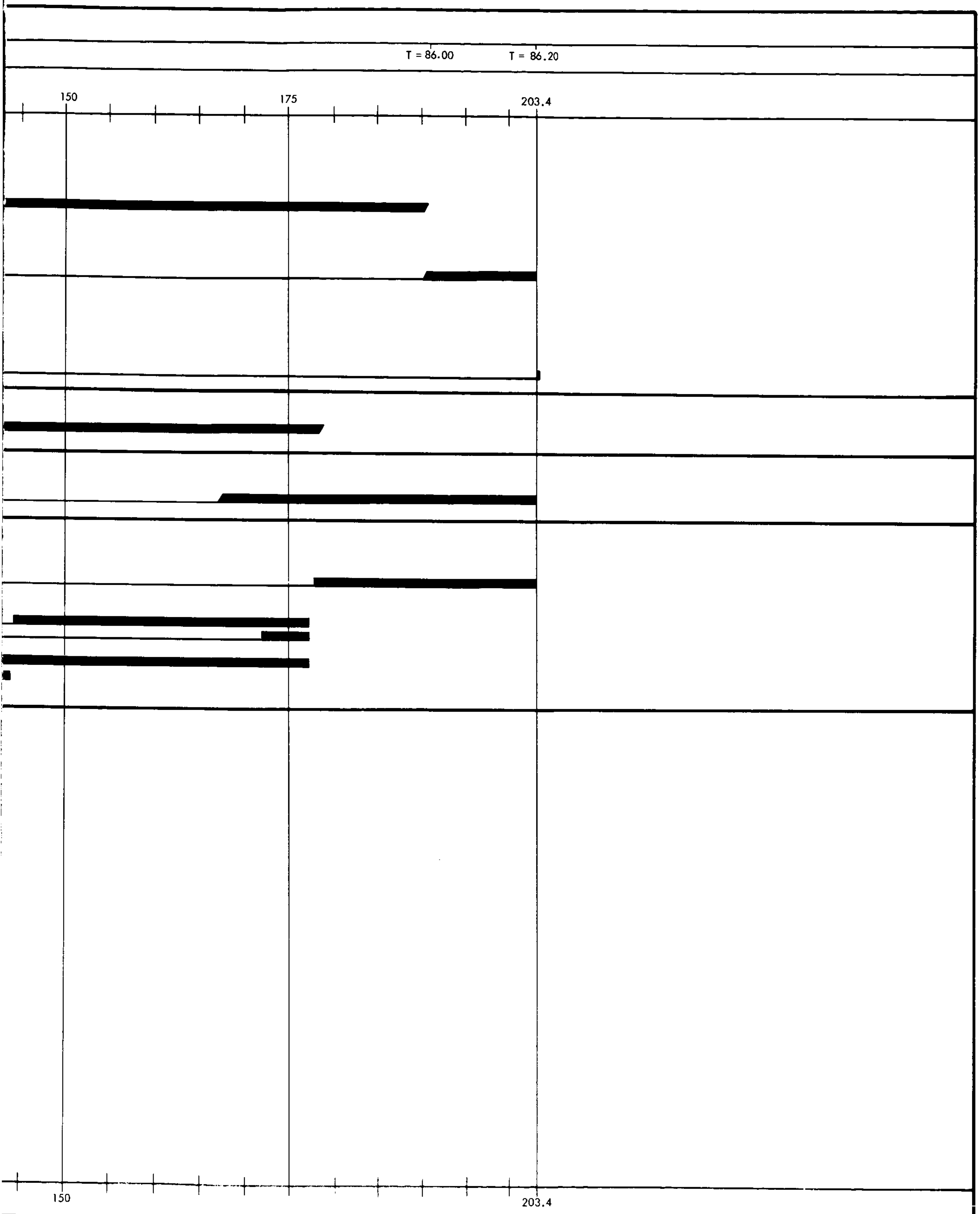
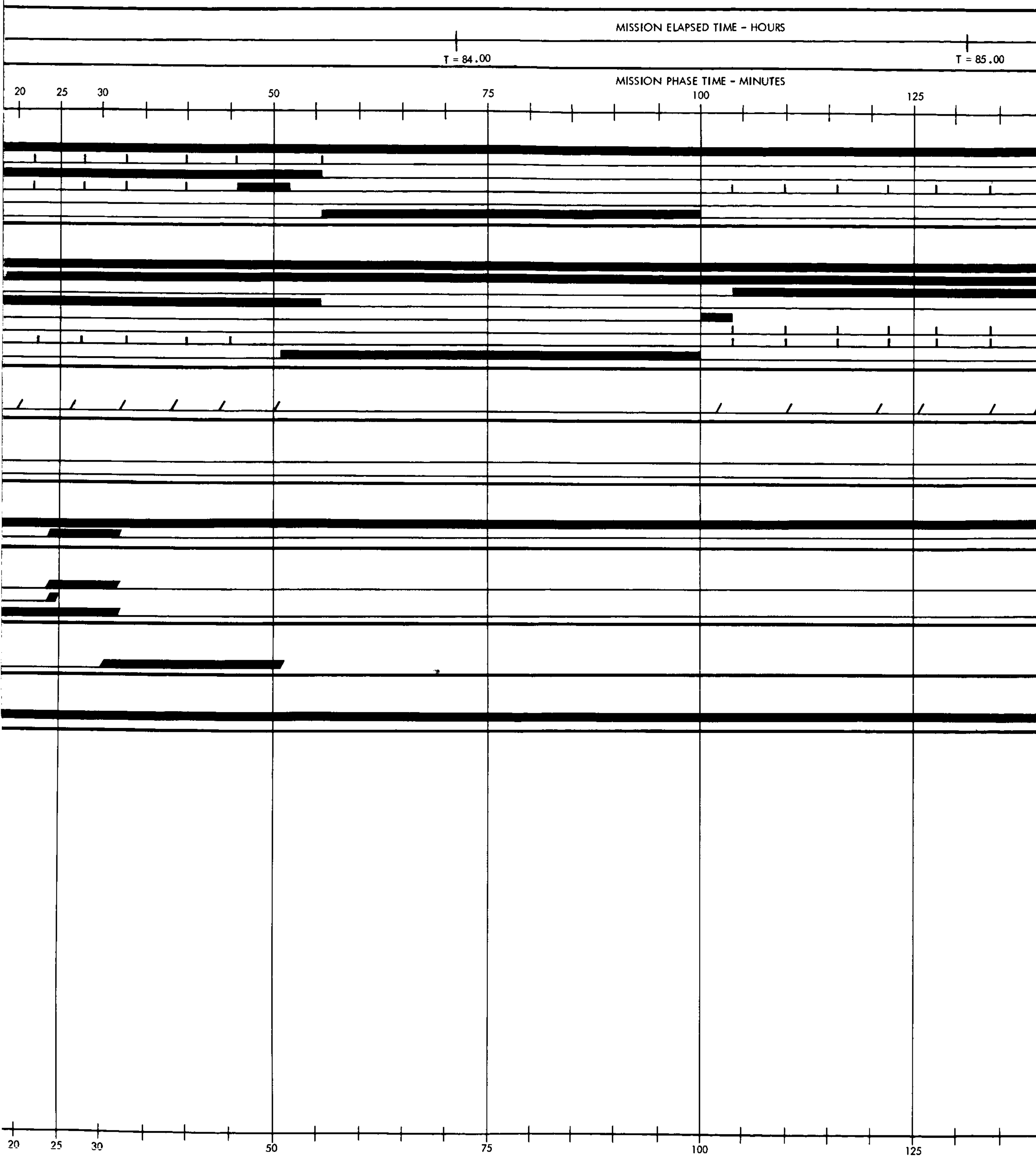


Figure 37. Mission Phase Time Line - Lunar Orbit (Subsequent to LEM Rendezvous) (Sheet 1 of 2)

[redacted]

[redacted]

PERTINENT FUNCTIONS	T = 82.81 T=83.00 0 5 10 15
GUIDANCE AND NAVIGATION SYSTEM	
PRIMARY INERTIAL REFERENCE	
CONTROLLED ROTATION TO SPECIFIED ATTITUDES	
G AND N ATTITUDE HOLD MODE	
LUNAR ORBIT AND EPHEMERIDES	
TRANSEARTH INJECTION PARAMETERS	
SCS MONITOR MODE	
STABILIZATION AND CONTROL SYSTEM	
SECONDARY INERTIAL REFERENCE	
ATTITUDE RATE-OF-CHANGE	
SCS ATTITUDE HOLD MODE	
G AND N ATTITUDE HOLD MODE	
SCS LOCAL VERTICAL MODE	
CONTROLLED ROTATION TO SPECIFIED ATTITUDES	
FREE DRIFT OR FREE ROTATION AROUND AN AXIS	
SCS MONITOR MODE	
S/M REACTION CONTROL SYSTEM	
ATTITUDE & TRANSLATION IMPULSES	
SERVICE PROPULSION SYSTEM	
GIMBAL OPERATION & ANGLE PRESETTING	
PROPELLANT UTILIZATION & FLOW RATIO ADJUSTMENT	
ENVIRONMENTAL CONTROL SYSTEM	
"SHIRT SLEEVE" ENVIRONMENT	
PRESSURE SUIT ENVIRONMENT	
CREW EQUIPMENT SYSTEM	
CREW SUPPORT & RESTRAINT	
REPLACE CENTER COUCH	
PRESSURE SUIT ENVIRONMENT	
IN-FLIGHT TEST SYSTEM	
CRITICAL SYSTEMS CHECKOUT	
ELECTRICAL POWER SYSTEM	
MAIN POWER - AC & DC	
	0 5 10 15


MISSION ELAPSED TIME - HOURS
T = 84.00
T = 85.00
MISSION PHASE TIME - MINUTES
20
25
30
50
75
100
125
20
25
30
50
75
100
125

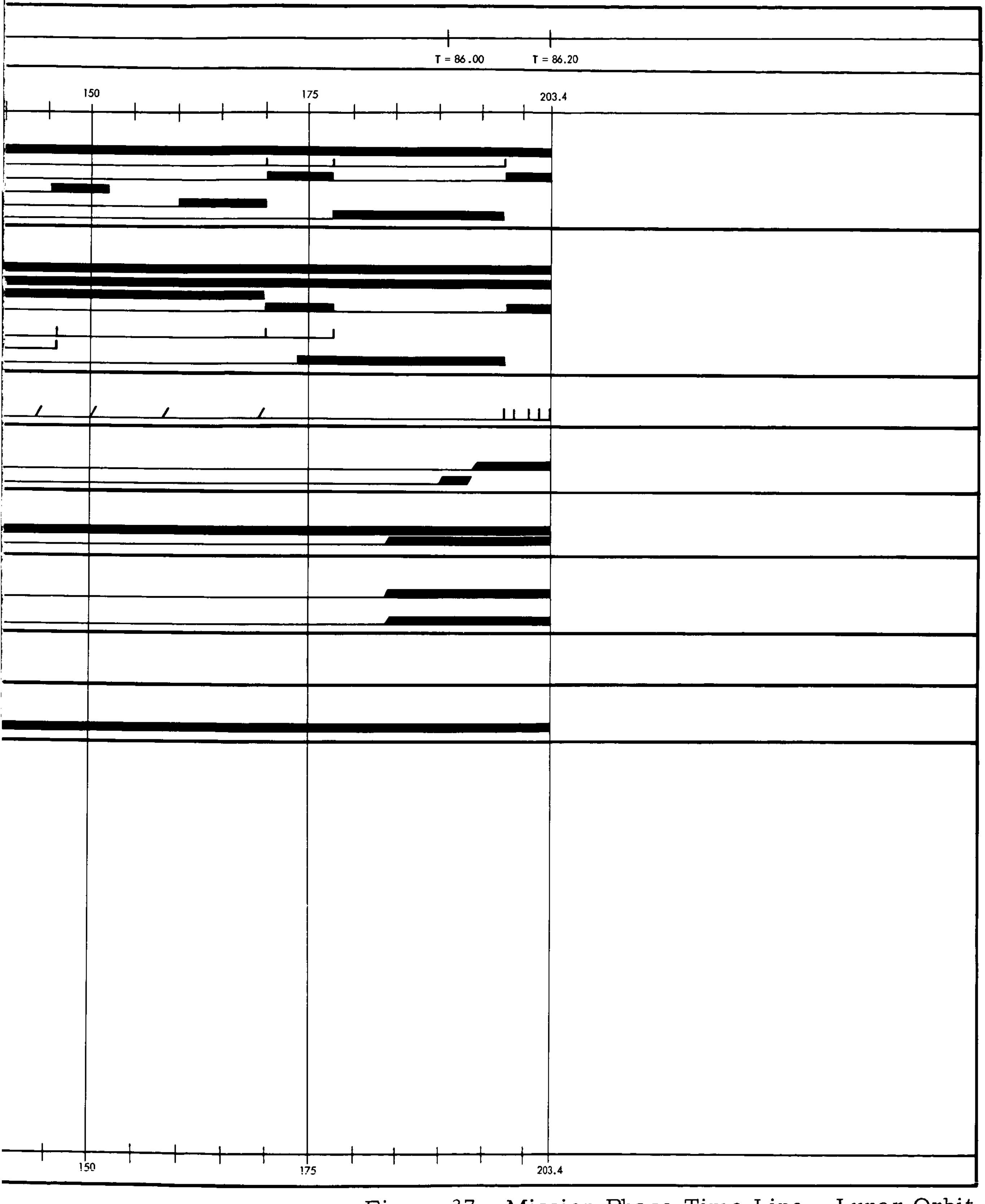


Figure 37. Mission Phase Time Line - Lunar Orbit (Subsequent to LEM Rendezvous) (Sheet 2 of 2)

[redacted]

[redacted]

TRANSEARTH INJECTION PHASE

The Transearth Injection Phase begins with S/M Reaction Control System ullage acceleration and ends with Service Propulsion System cutoff.

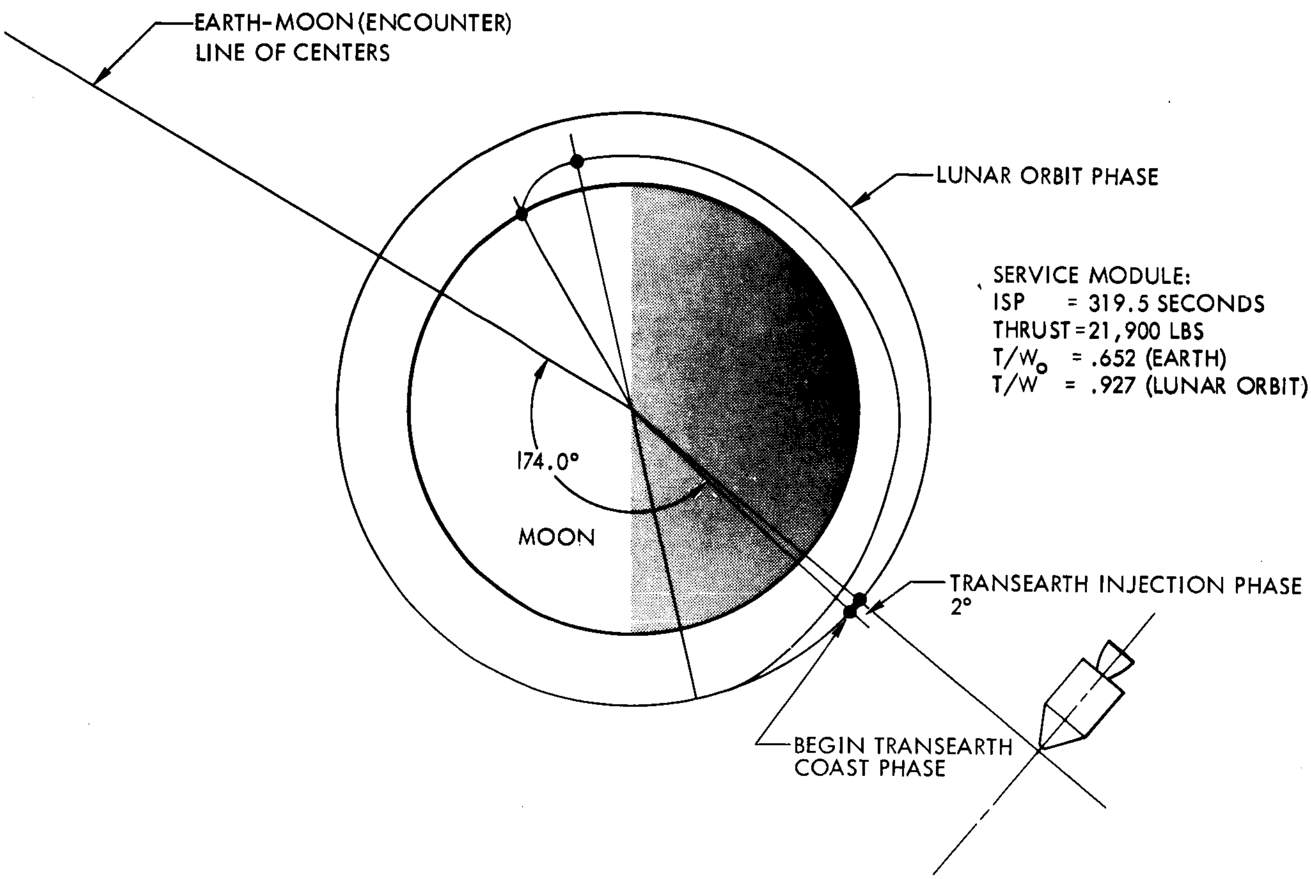
Figure 38 describes the geometry of the Transearth Injection Phase.

Figure 39 is a two-page time-line delineation of spacecraft system activity during the Treansearth Injection Phase.

[redacted]

MISSION EVENTS

(1) S/M RCS ULLAGE ACCELERATION → (2) SERVICE PROPULSION SYSTEM IGNITION → (3) SERVICE PROPULSION SYSTEM CUTOFF



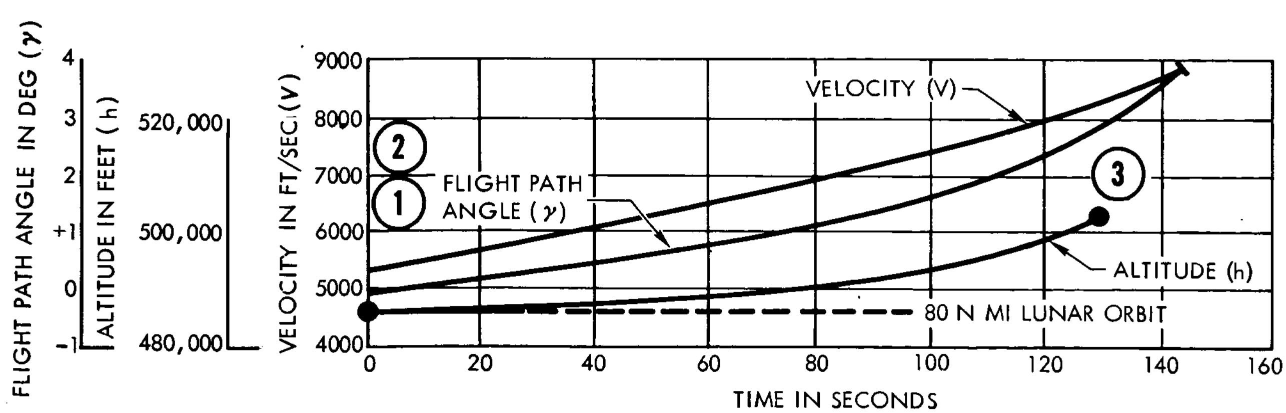


Figure 38. Transearth Injection Phase

[redacted]

[redacted]

MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION
S/C ULLAGE ACCELERATION.	86.20	T = 1 SEC
S/C SPS IGNITION & OPERATION.		T = 127.5 SEC
G & N PROGRAMMED MANEUVER.		T = 127.5 SEC
S/C SPS CUTOFF.	86.24	

T = 86.20

0 5 10

POSITIONAL DATA — ESTIMATED

S/C ON OPPOSITE SIDE OF MOON FROM EARTH

PERTINENT FUNCTIONS

COMMUNICATIONS & INSTRUMENTATION SYSTEM

DATA STORAGE RECORDING

0 5 10

MISSION ELAPSED TIME - HOURS

MISSION PHASE TIME - SECONDS

20 30 40 50 60 70

20 30 40 50 60 70

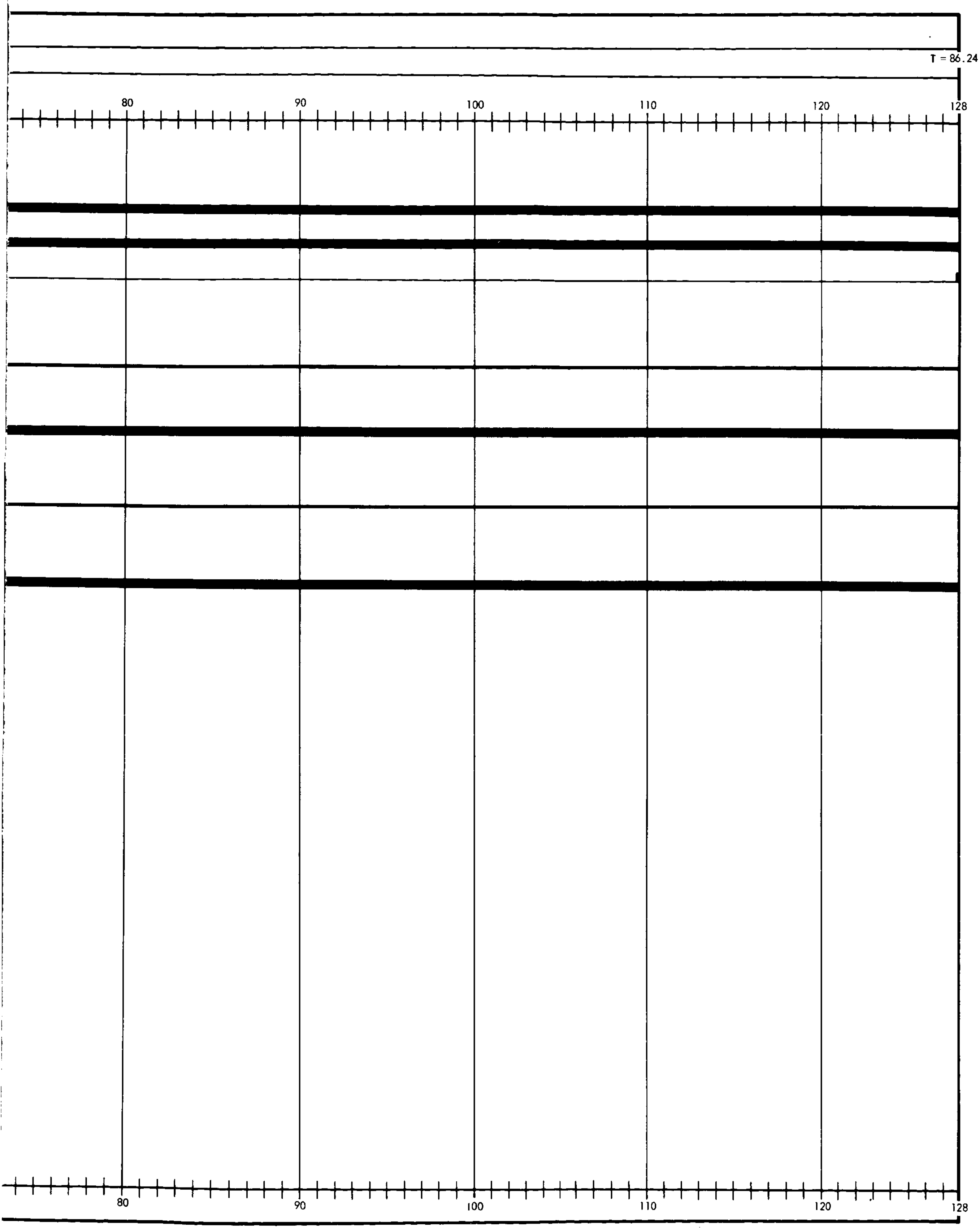


Figure 39. Mission Phase Time Line - Transearth Injection (Sheet 1 of 2)

[redacted]

PERTINENT FUNCTIONS

T = 86.20

0 5 10

GUIDANCE AND NAVIGATION SYSTEM

PRIMARY INERTIAL REFERENCE.

G AND N LARGE ΔV MODE.

STABILIZATION AND CONTROL SYSTEM

SECONDARY INERTIAL REFERENCE

ATTITUDE RATE-OF-CHANGE

G AND N LARGE Δ V MODE

X - AXIS VELOCITY DATA

TIME DATA

S/M REACTION CONTROL SYSTEM

TRANSLATION & ATTITUDE IMPULSES

SERVICE PROPULSION SYSTEM

THRUST IMPULSE

GIMBAL OPERATION

ENVIRONMENTAL CONTROL SYSTEM

PRESSURE SUIT ENVIRONMENT

CREW EQUIPMENT SYSTEM

CREW SUPPORT & RESTRAINT

PRESSURE SUIT ENVIRONMENT

ELECTRICAL POWER SYSTEM

MAIN POWER - AC & DC

0 5 10

MISSION ELAPSED TIME - HOURS

MISSION PHASE TIME - SECONDS

20 30 40 50 60 70

20 30 40 50 60 70

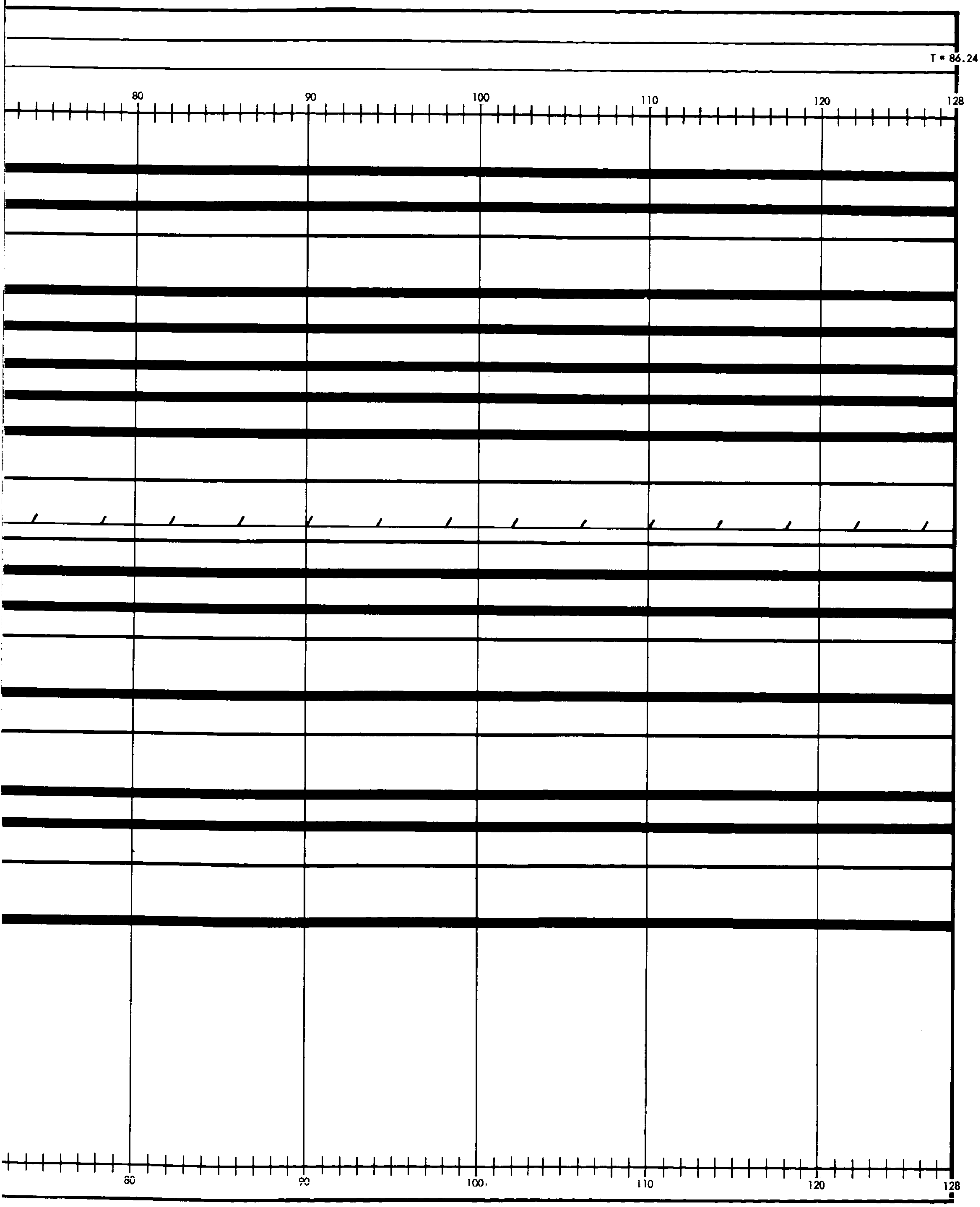


Figure 39. Mission Phase Time Line - Transearth Injection (Sheet 2 of 2)

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TRANSEARTH COAST PHASE

The Transearth Coast Phase begins with Service Propulsion System cutoff and ends when the Command Module begins entry into the earth's atmosphere (400,000 ft.).

Figure 40 describes the geometry of the Translunar Coast Phase.

Figure 41 is an earth trace of the Translunar Coast Phase superimposed on a trace for the entire mission.

Figure 42 is a two-page time-line delineation of spacecraft system activity during the Transearth Coast Phase.

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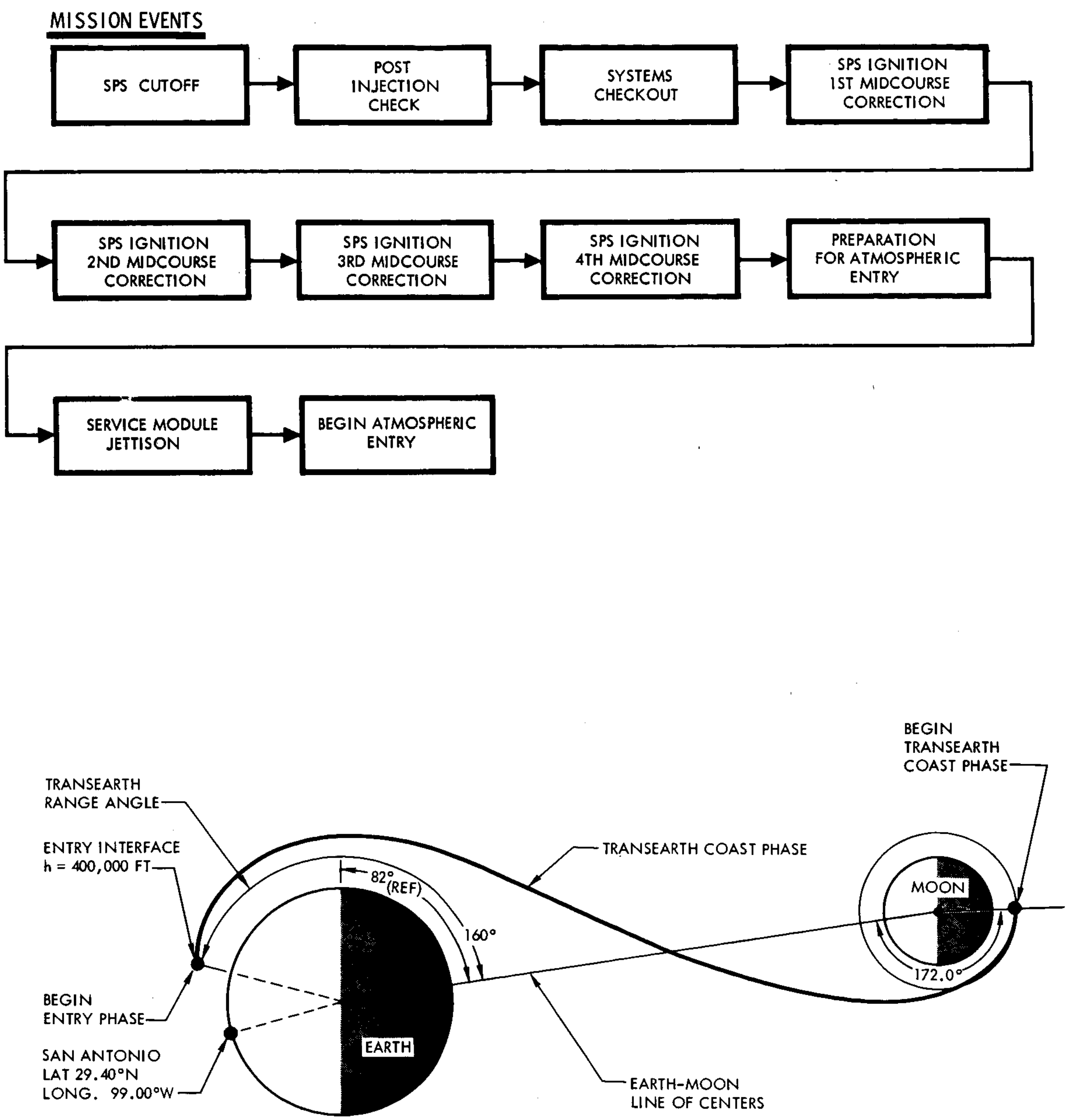
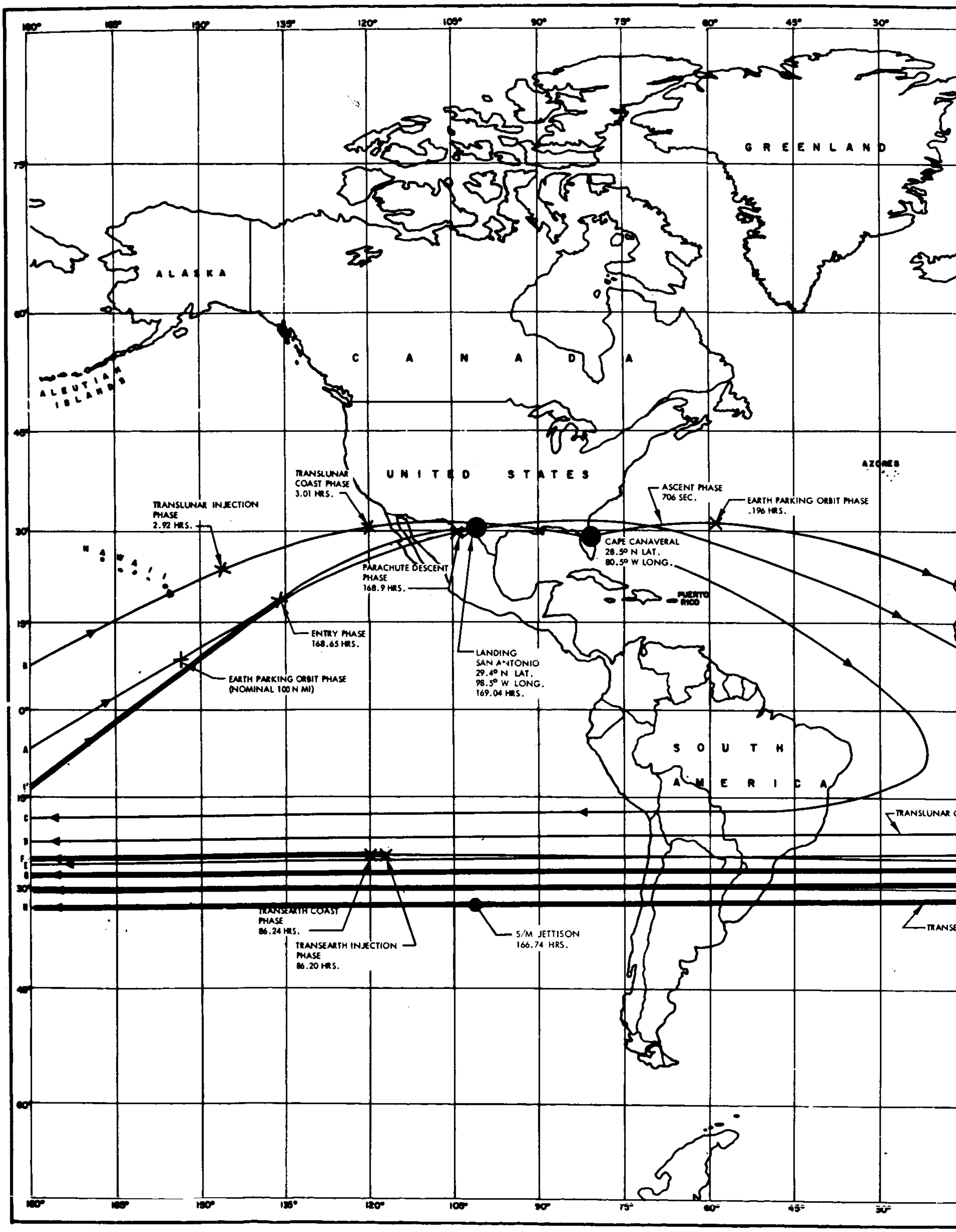


Figure 40. Transearth Coast Phase

~~CONFIDENTIAL~~

[redacted]



ALASKA
ALEUTIAN ISLANDS
CANADA
GREENLAND
UNITED STATES
HAWAII
AZORES
PUERTO RICO
SOUTH AMERICA
TRANSLUNAR INJECTION PHASE 2.92 HRS.
TRANSLUNAR COAST PHASE 3.01 HRS.
PARACHUTE DESCENT PHASE 168.9 HRS.
ASCENT PHASE 706 SEC.
EARTH PARKING ORBIT PHASE .196 HRS.
CAPE CANAVERAL 28.5° N LAT. 80.5° W LONG.
ENTRY PHASE 168.65 HRS.
EARTH PARKING ORBIT PHASE (NOMINAL 100 N MI)
LANDING SAN ANTONIO 29.4° N LAT. 98.5° W LONG. 169.04 HRS.
TRANSLUNAR C
TRANSEARTH COAST PHASE 86.24 HRS.
TRANSEARTH INJECTION PHASE 86.20 HRS.
S/M JETTISON 166.74 HRS.
TRANSE

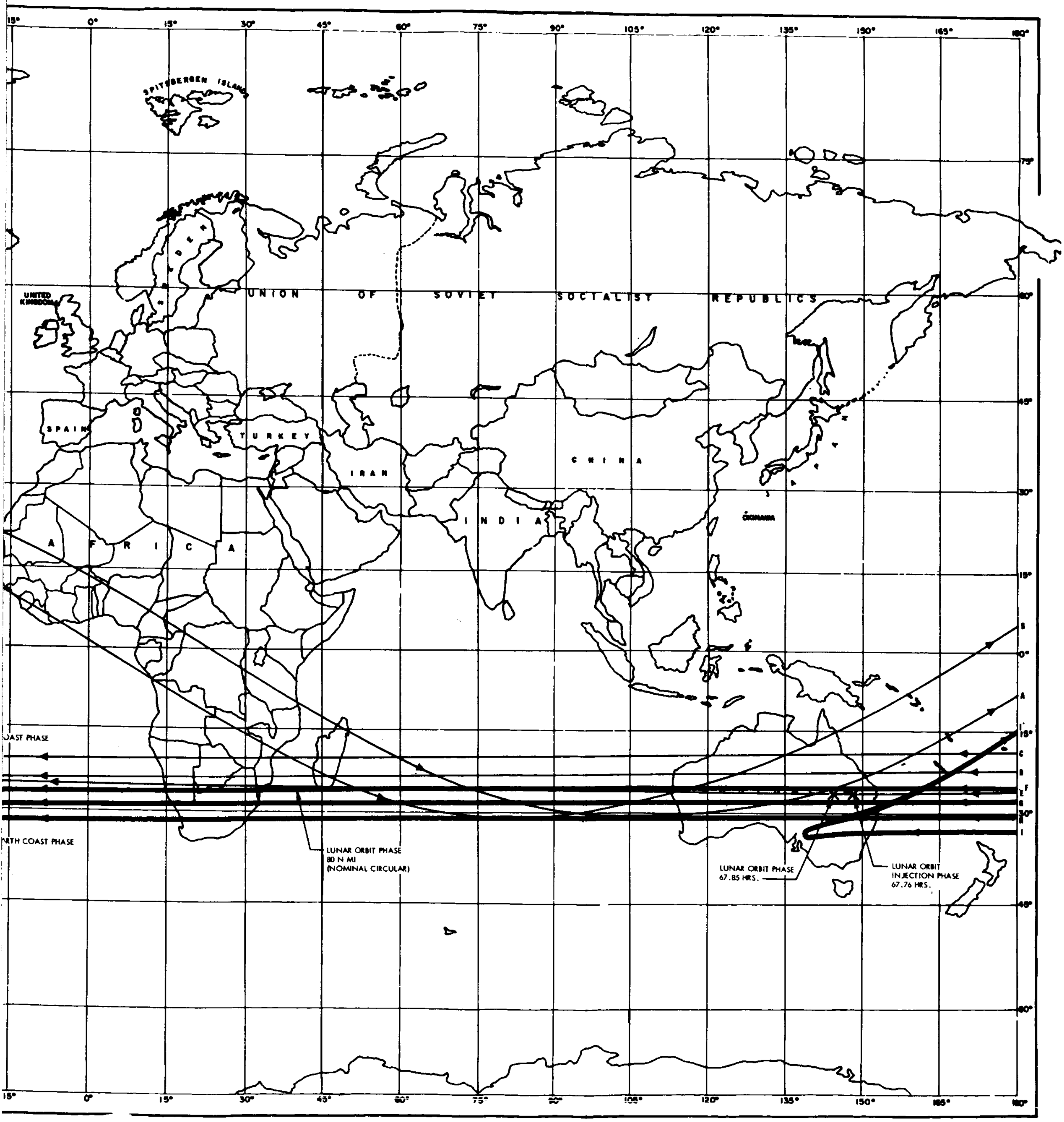


Figure 41. Mission Trajectory Earth Trace-Transearth Coast

 [redacted]

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MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION
SPS CUTOFF	T = 86.24	
S/C ATTITUDE STABILIZATION & CONTROL PERIODS		
POST INJECTION CHECK & COAST ARRANGEMENT		15 MIN
CHECKLIST VERIFICATION OF CONTROL SETTINGS		
CHECKLIST VERIFICATION OF INSTRUMENT READINGS		
CREW & EQUIP. ARRANGEMENT FOR COAST		
OPERATIONAL CAPABILITY VERIFICATION		30 MIN
IFTS & INSTRUMENTATION CHECKS		
1ST MIDCOURSE CORRECTION	T = 86.74	5 HRS
MANUAL TRAJECTORY DATA (10 SIGHTINGS AT 1/2 HR.)		2 MIN
TRAJECTORY ERROR PARAMETERS COMPUTATION		15 MIN
IMU FINE ALIGNMENT		10 MIN
COUNTDOWN (SYST. SET-UP, SECURING, ORIENTATION)		10 MIN
VELOCITY/VECTOR CHANGE		1 SEC
POST SPS IMPULSE CHECK (CONTROLS, READOUTS, ARRANGE)		5 MIN
2ND MIDCOURSE CORRECTION (REFER TO 1ST MIDCOURSE CORRECTION)	T = 91.74	5 HRS
3RD MIDCOURSE CORRECTION (SIMILAR TO 1ST MIDCOURSE CORRECTION EXCEPT THE 10 SIGHTINGS ARE TAKEN AT APPROX 3 HR. INTERVALS)	T = 98.00	30 HRS
4TH MIDCOURSE CORRECTION (SIMILAR TO 1ST MIDCOURSE CORRECTION EXCEPT THE 10 SIGHTINGS ARE TAKEN AT APPROX. 1 HR. INTERVALS)	T = 154.00	10 HRS
EARTH ENTRY PREPARATION	T = 164.65	4 HRS
EARTH ENTRY PARAMETERS COMPUTATION		30 MIN
C/M SET-UP FOR S/M JETTISON		30 MIN
S/M JETTISON		DISCRETE
C/M SYSTEMS SET-UP FOR ENTRY		10 MIN
CREW & EQUIPMENT ARRANGEMENT		10 MIN
ENTRY ORIENTATION		10 MIN
400,000 FT. ALTITUDE PENETRATION & START ENTRY	T = 168.65	

GOSS COVERAGE ESTIMATED

GOLDSTONE DSIF
JOHANNESBURG DSIF
WOOMERA DSIF
CANTON
KAUAI

POSITIONAL DATA-ESTIMATED

S/C IN VAN ALLEN RADIATION BELT

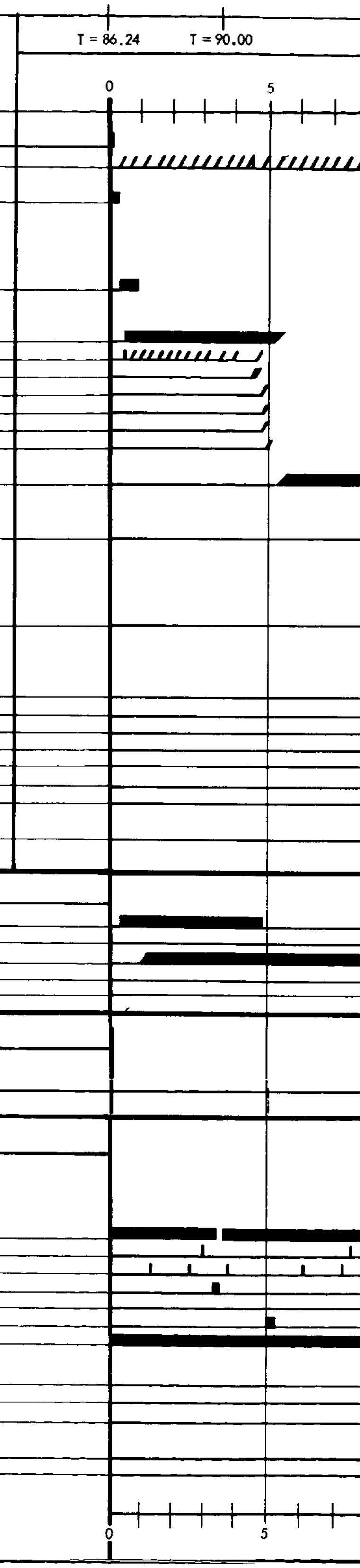
PERTINENT FUNCTIONS

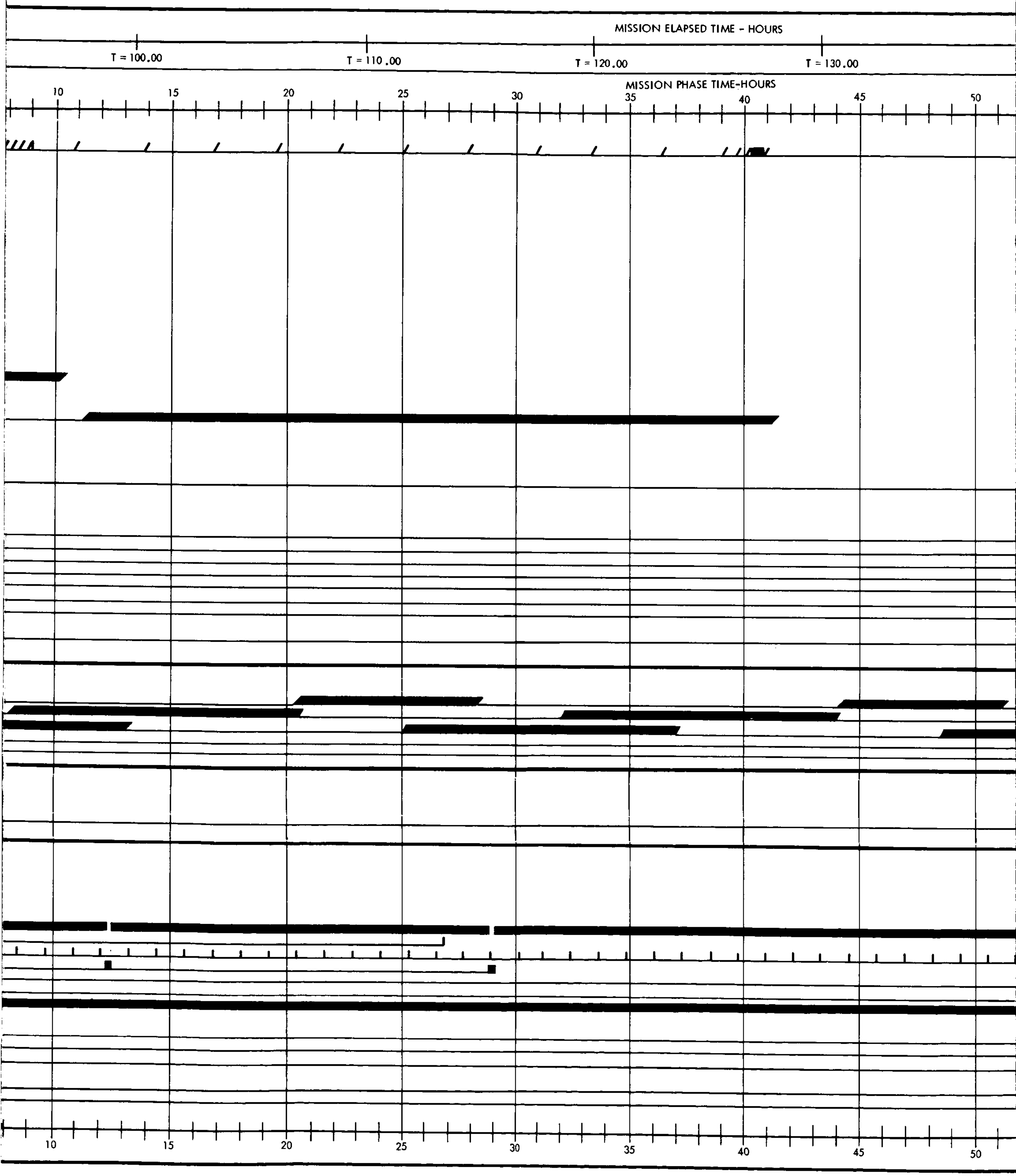
COMMUNICATIONS & INSTRUMENTATION SYSTEM

DSIF NARROW BAND TELEMETRY
DATA STORAGE RECORDING
DSIF 2-WAY VOICE WITH GOSS
DSIF DATA STORAGE TRANSMISSION
NEAR EARTH 2-WAY DOPPLER TRACKING OR RANGING
C/M DSIF TV TRANSMISSION
DSIF 2-WAY DOPPLER TRACKING OR RANGING

MAIN ANTENNA INOPERABLE:
DSIF 2-WAY VOICE WITH GOSS
DSIF 2-WAY DOPPLER TRACKING OR RANGING
DSIF NARROW BAND TELEMETRY

NEAR EARTH TELEMETRY
NEAR EARTH 2-WAY VOICE WITH GOSS




MISSION ELAPSED TIME - HOURS
T = 100.00
T = 110.00
T = 120.00
T = 130.00
MISSION PHASE TIME-HOURS
10
15
20
25
30
35
40
45
50

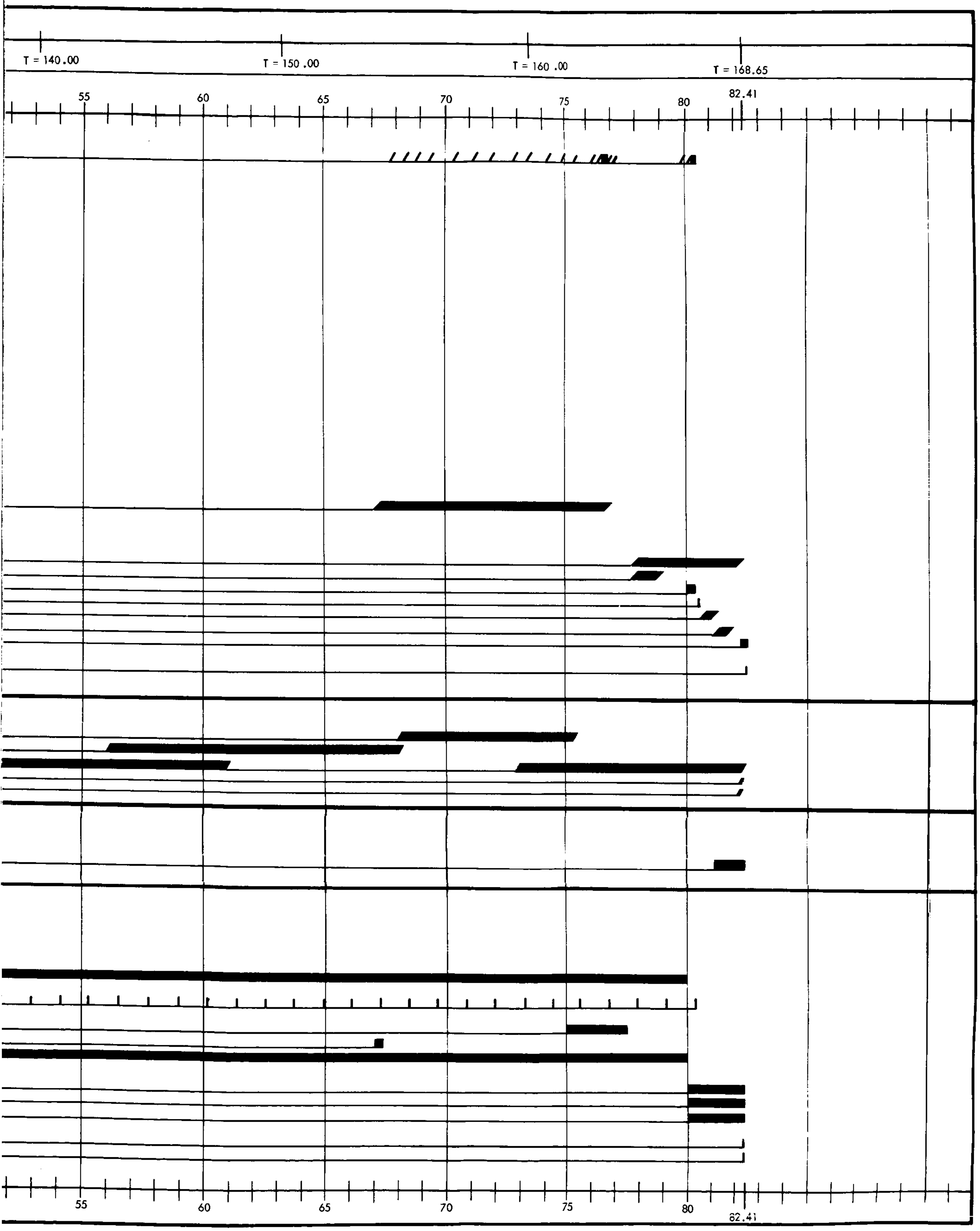


Figure 42. Mission Phase Time Line - Transearth Coast (Sheet 1 of 2)

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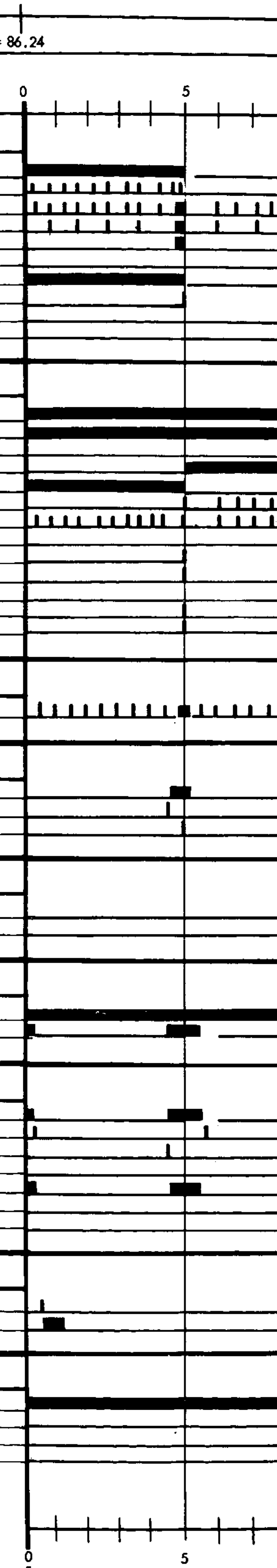
PERTINENT FUNCTIONS



GUIDANCE AND NAVIGATION SYSTEM

PRIMARY INERTIAL REFERENCE
CONTROLLED ROTATION TO SPECIFIED ATTITUDE
PRESENT TRANSEARTH TRAJECTORY
TRANSEARTH TRAJECTORY MISS-DISTANCE
TRANSEARTH MIDCOURSE CORRECTION PARAMETERS
SCS MONITOR MODE
G AND N ATTITUDE HOLD MODE
G AND N LARGE Δ V MODE
ON-OFF THRUST SIGNALS FOR G AND N SMALL ΔV.
EARTH ENTRY PARAMETERS

STABILIZATION AND CONTROL SYSTEM

SECONDARY INERTIAL REFERENCE
ATTITUDE RATE-OF-CHANGE
SCS MONITOR MODE
SCS ATTITUDE HOLD MODE
G AND N ATTITUDE HOLD MODE
CONTROLLED ROTATION TO SPECIFIED ATTITUDE
FREE DRIFT OR FREE ROTATION AROUND AN AXIS
SCS LARGE ΔV MODE
G AND N LARGE ΔV MODE
SCS SMALL TRANSLATION THRUST
ON-OFF THRUST SIGNALS FOR G AND N SMALL ΔV DISPLAY
X-AXIS VELOCITY DATA
TIME DATA

S/M - REACTION CONTROL SYSTEM

ATTITUDE & TRANSLATION IMPULSES

SERVICE PROPULSION SYSTEM

GIMBAL OPERATION & ANGLE PRESETTING
PROPELLANT UTILIZATION & FLOW RATIO ADJUSTMENT
THRUST IMPULSE

C/M - REACTION CONTROL SYSTEM

INITIAL PRESSURIZATION SEQUENCE
ATTITUDE IMPULSES

ENVIRONMENTAL CONTROL SYSTEM

"SHIRT SLEEVE" ENVIRONMENT
PRESSURE SUIT ENVIRONMENT

CREW EQUIPMENT SYSTEM

CREW SUPPORT & RESTRAINT
REPOSITION CENTER COUCH
REPLACE CENTER COUCH
HYGIENE & HEALTH FUNCTION
PRESSURE SUIT ENVIRONMENT
WASTE MANAGEMENT
FOOD MANAGEMENT

IN-FLIGHT TEST SYSTEM

AUTOMATIC SYSTEMS CHECKOUT
MANUAL SYSTEMS CHECKOUT

ELECTRICAL POWER SYSTEM

MAIN POWER - AC & DC
ENTRY BATTERY RECHARGING
POST LANDING BATTERY RECHARGING
ENTRY POWER - AC & DC


MISSION ELAPSED TIME - HOURS
T = 110.00
T = 120.00
T = 130.00
MISSION PHASE TIME - HOURS
10
15
20
25
30
35
40
45
50

Figure 42. Mission Phase Time Line - Transearth Coast (Sheet 2 of 2)

ENTRY PHASE

The Entry Phase begins with atmospheric entry at 400,000 ft. and ends with drogue chute deployment at 40,000 ft.

Figure 43 describes the geometry of the Entry Phase.

Figure 44 is an earth trace of the Entry Phase superimposed on a trace for the entire mission.

Figure 45 is a two-page time-line delineation of spacecraft system activity during the Entry Phase.

[redacted]

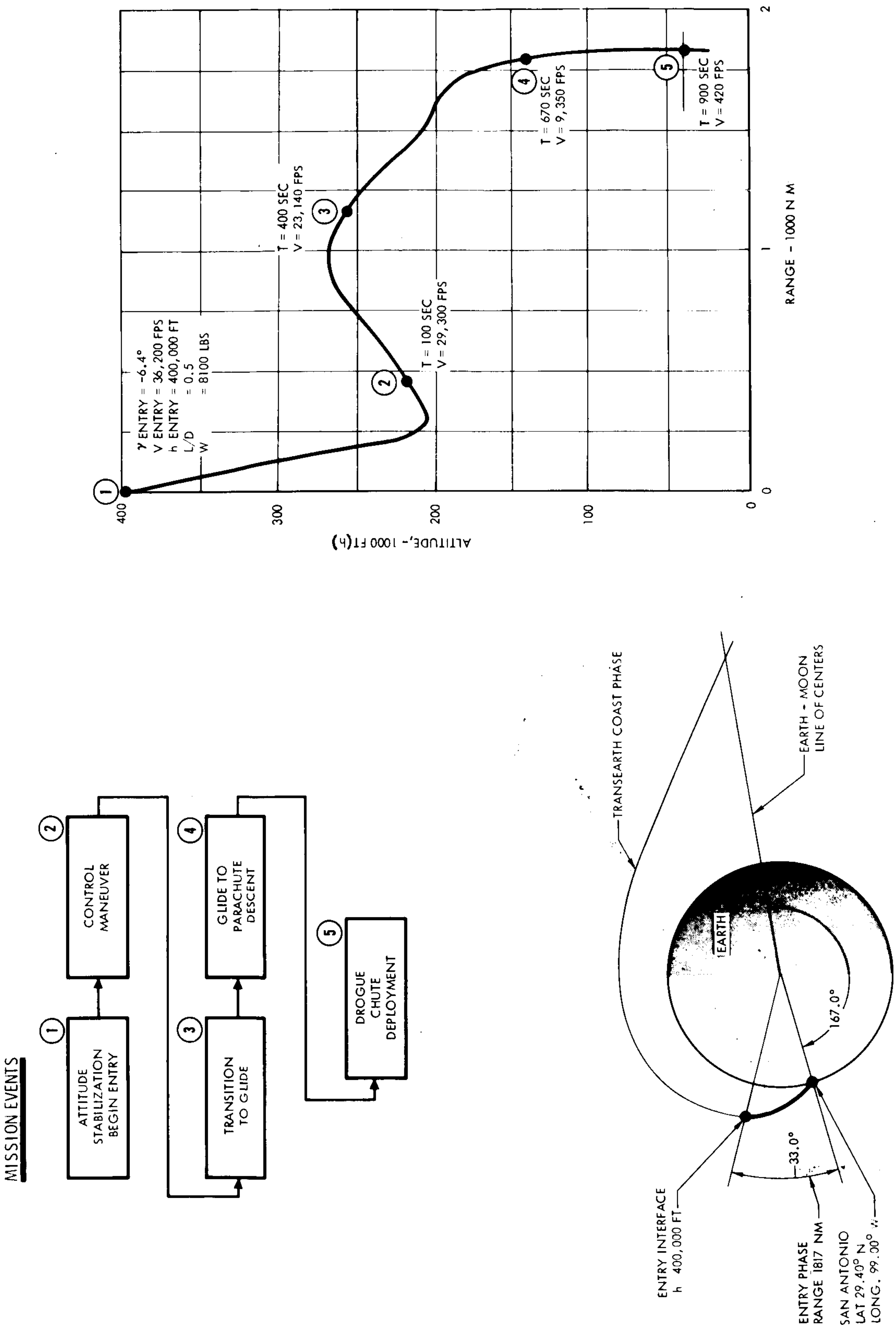
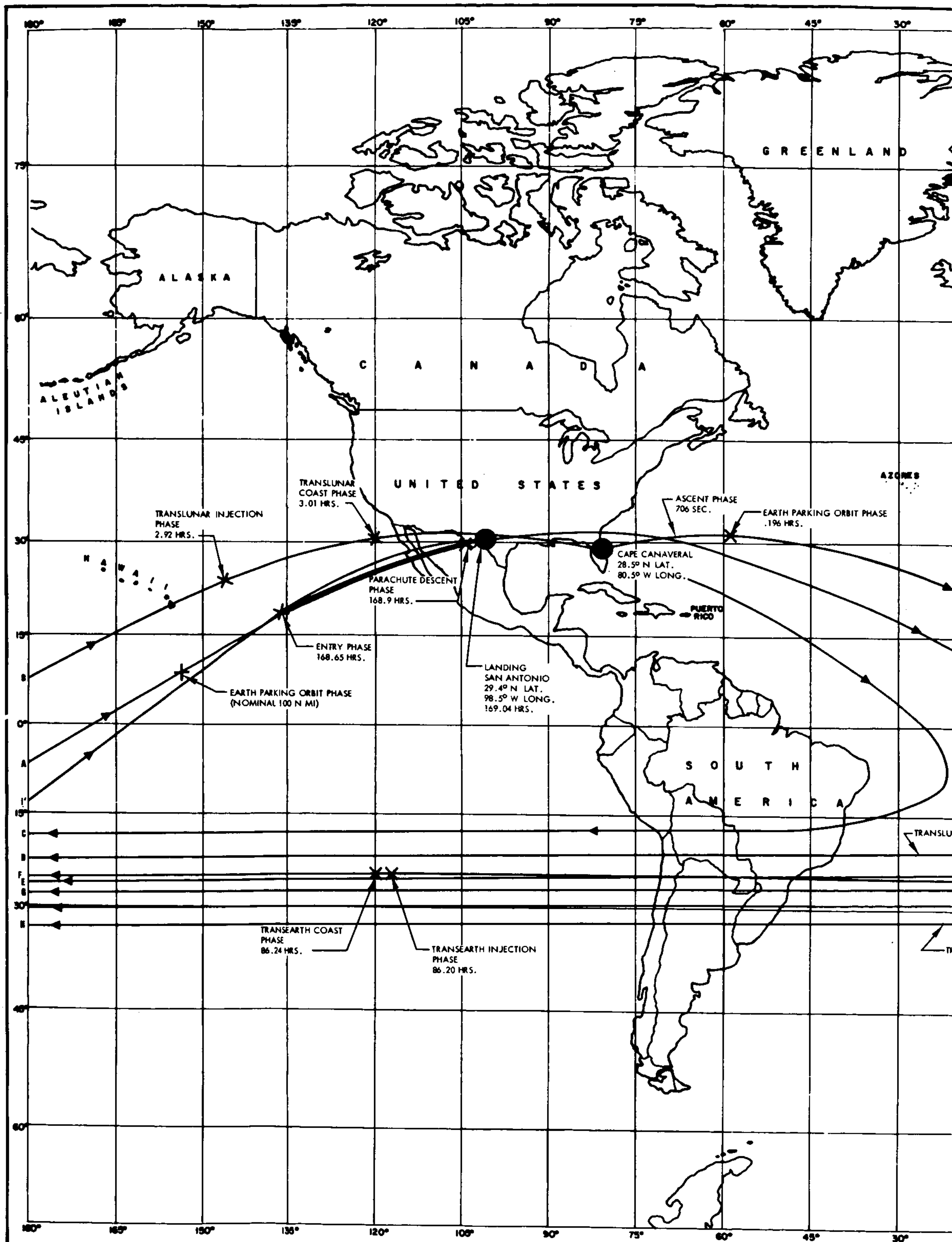


Figure 43. Entry Phase

~~CONFIDENTIAL~~



MISSION TRAJECTORY - EARTH TRACE

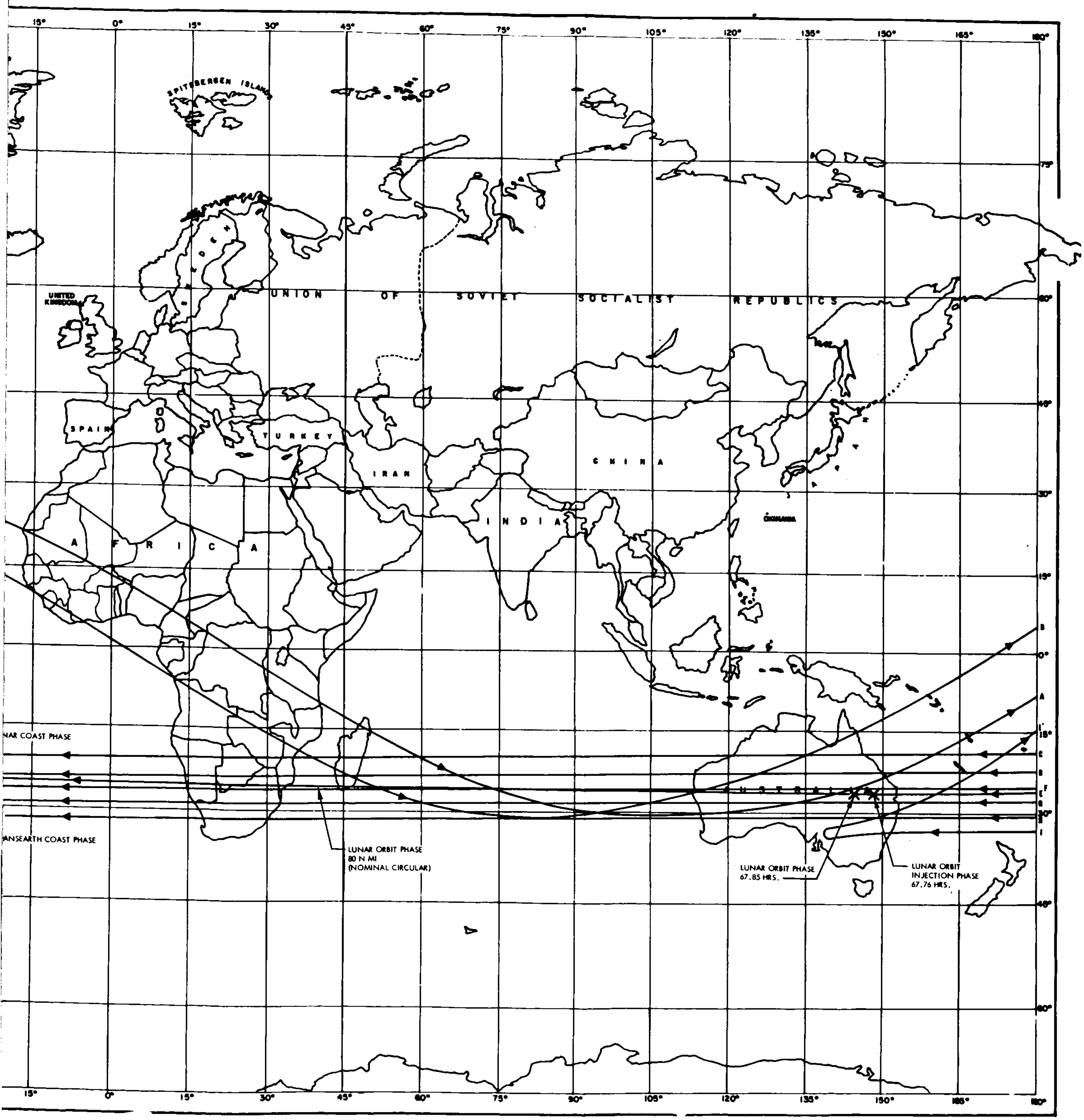


Figure 44. Mission Trajectory Earth Trace-Entry

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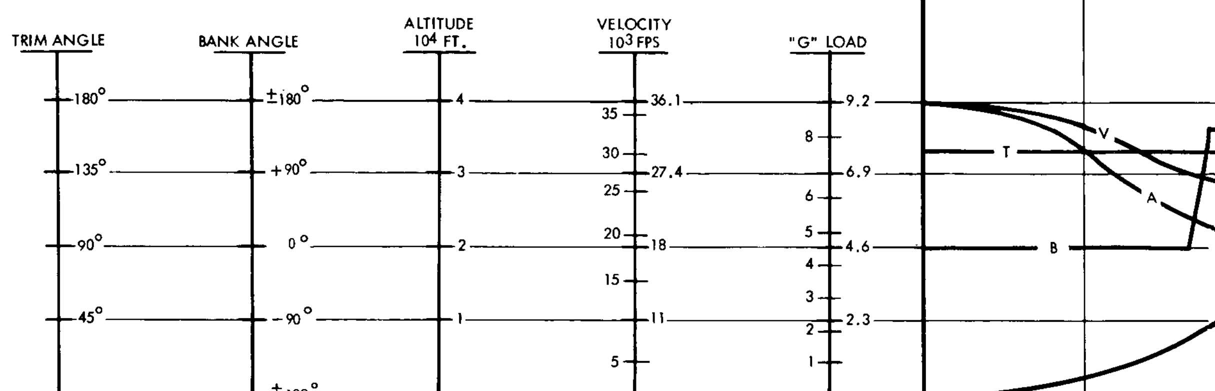
MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HRS.	EVENT DURATION	T = 168.65
ATTITUDE STABILIZATION	T = 168.65		
INITIAL MANEUVERS FOR MAIN CONTROL OF ENTRY RANGE CONTROL ROLL MANEUVERS TRAJECTORY PATH MONITORING & PREDICTING		300 SEC	
TRANSITION TO GLIDE SECONDARY MANEUVERS FOR MINOR PATH CONTROL		270 SEC	
FINAL GLIDE APPROACH		230 SEC	
FORWARD HEAT SHIELD JETTISON	T = 168.90		

0 10 50

PERTINENT DATA - ESTIMATED



GOSS COVERAGE — ESTIMATED

GUAYMAS

WHITE SANDS

SAN ANTONIO

POSITIONAL DATA — ESTIMATED

S/C IN SUNLIGHT

PERTINENT FUNCTIONS

COMMUNICATIONS & INSTRUMENTATION SYSTEM

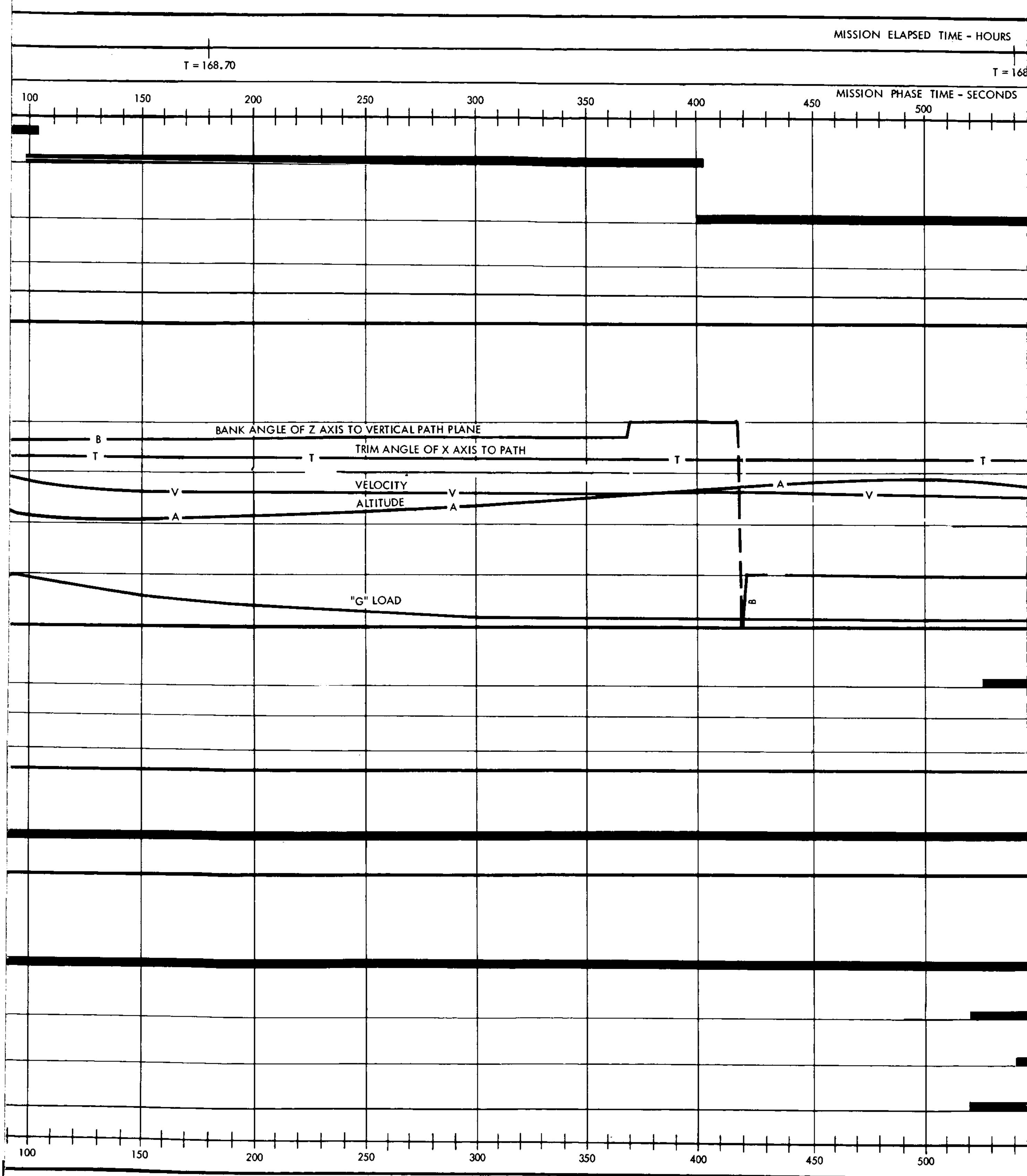
DATA STORAGE RECORDING

NEAR EARTH TELEMETRY

NEAR EARTH 2-WAY VOICE WITH GOSS

NEAR EARTH 2-WAY DOPPLER TRACKING OR RANGING

0 10 50


MISSION ELAPSED TIME - HOURS
T = 168.70
T = 168
MISSION PHASE TIME - SECONDS
100
150
200
250
300
350
400
450
500
BANK ANGLE OF Z AXIS TO VERTICAL PATH PLANE
B
TRIM ANGLE OF X AXIS TO PATH
T
VELOCITY
V
ALTITUDE
A
"G" LOAD

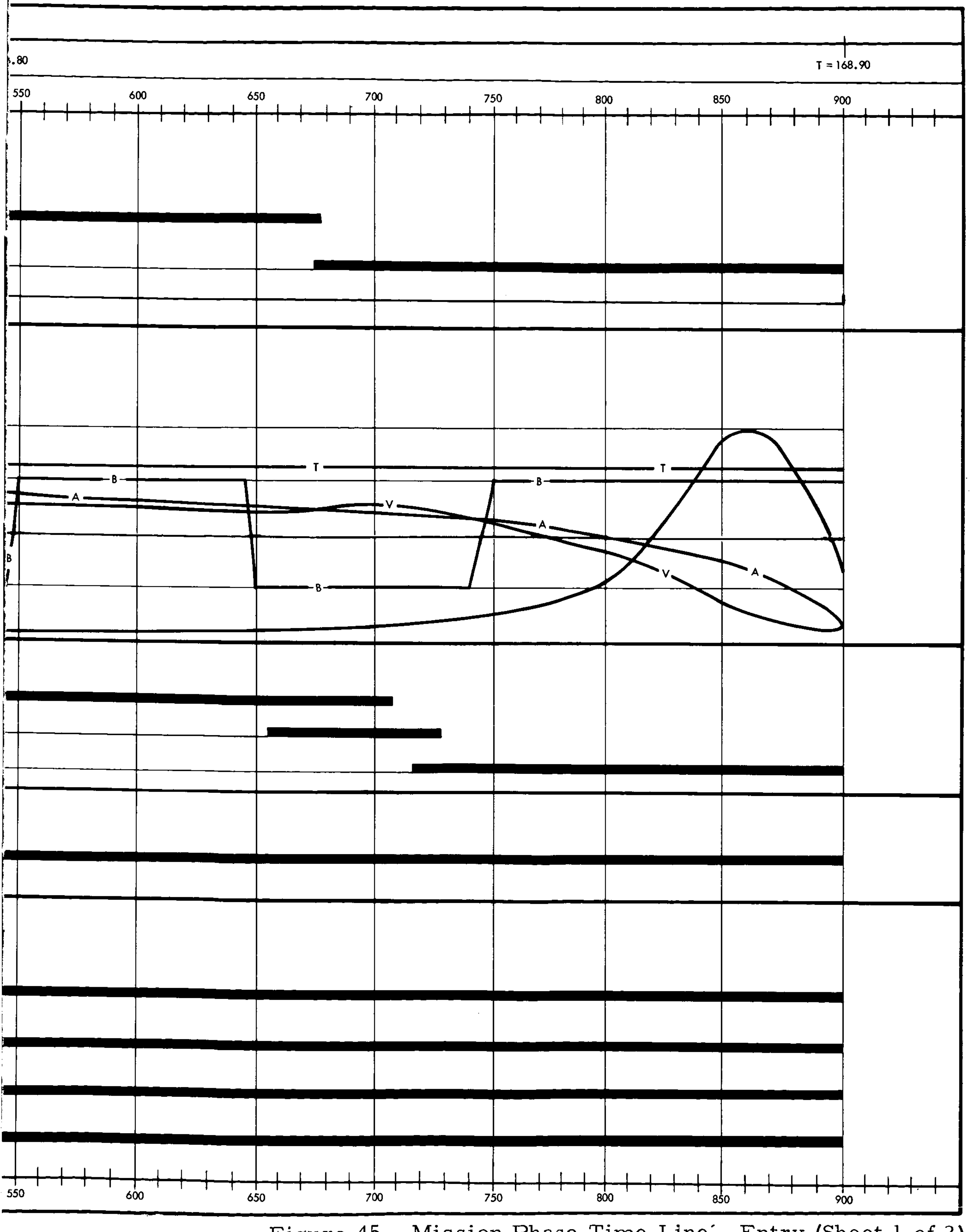


Figure 45. Mission Phase Time Line - Entry (Sheet 1 of 2)

~~CONFIDENTIAL~~

PERTINENT FUNCTONS

T= 168.65

0 10 50

GUIDANCE AND NAVIGATION SYSTEM

PRIMARY INERTIAL REFERENCE

CONTROLLED ROTATION TO SPECIFIED ATTITUDES

G AND N ENTRY MODE

STABILIZATION AND CONTROL SYSTEM

SECONDARY INERTIAL REFERENCE

ATTITUDE RATE-OF-CHANGE

X-AXIS VELOCITY DATA

TIME DATA

G AND N ENTRY MODE

C/M — REACTION CONTROL SYSTEM

ATTITUDE IMPULSES

ENVIRONMENTAL CONTROL SYSTEM

PRESSURE SUIT ENVIRONMENT

CREW EQUIPMENT SYSTEM

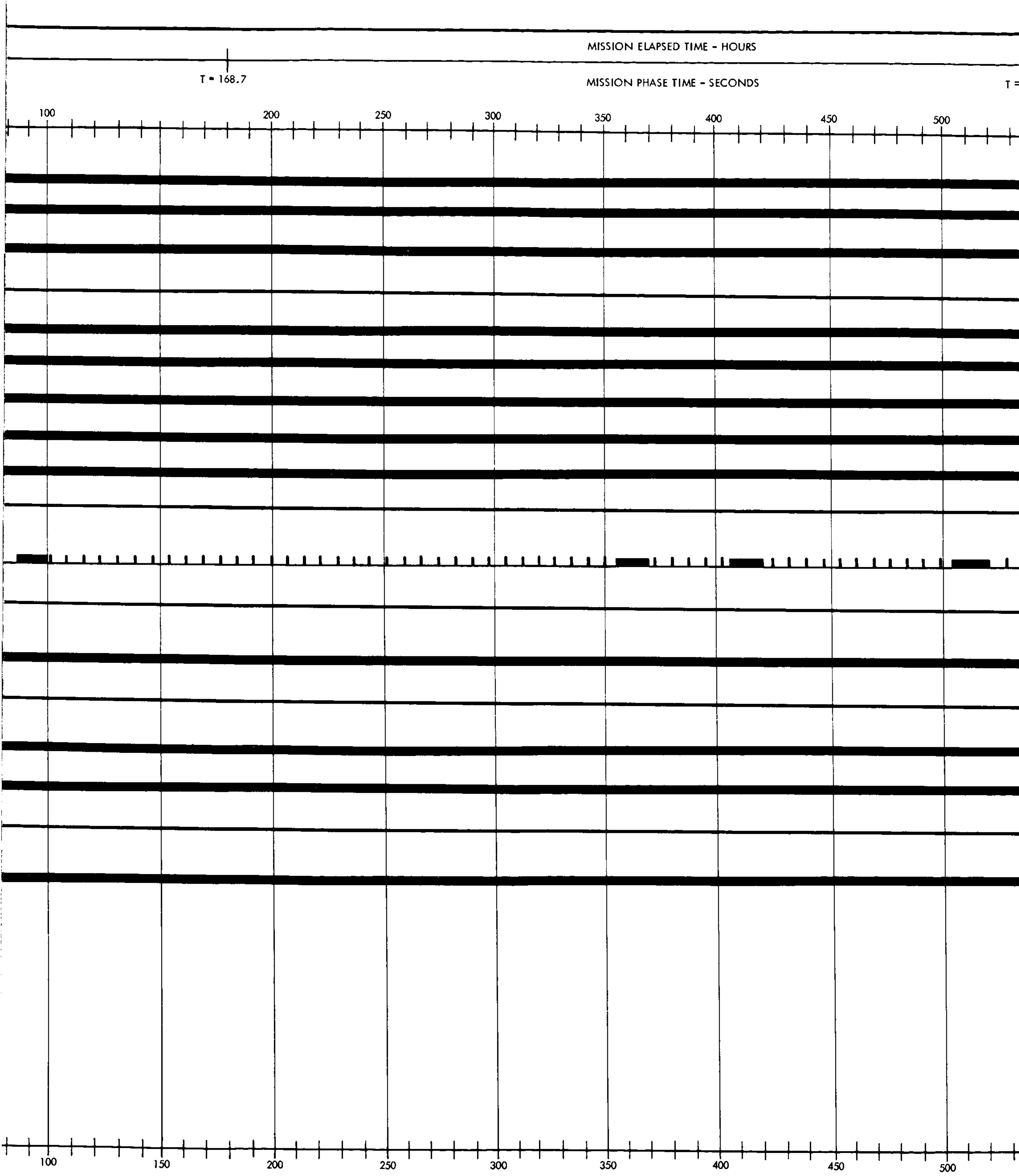
CREW SUPPORT & RESTRAINT.

PRESSURE SUIT ENVIRONMENT

ELECTRICAL POWER SYSTEM

ENTRY POWER AC & DC

0 10 50


MISSION ELAPSED TIME - HOURS
T = 168.7
MISSION PHASE TIME - SECONDS
T =
100
200
250
300
350
400
450
500
100
150
200
250
300
350
400
450
500

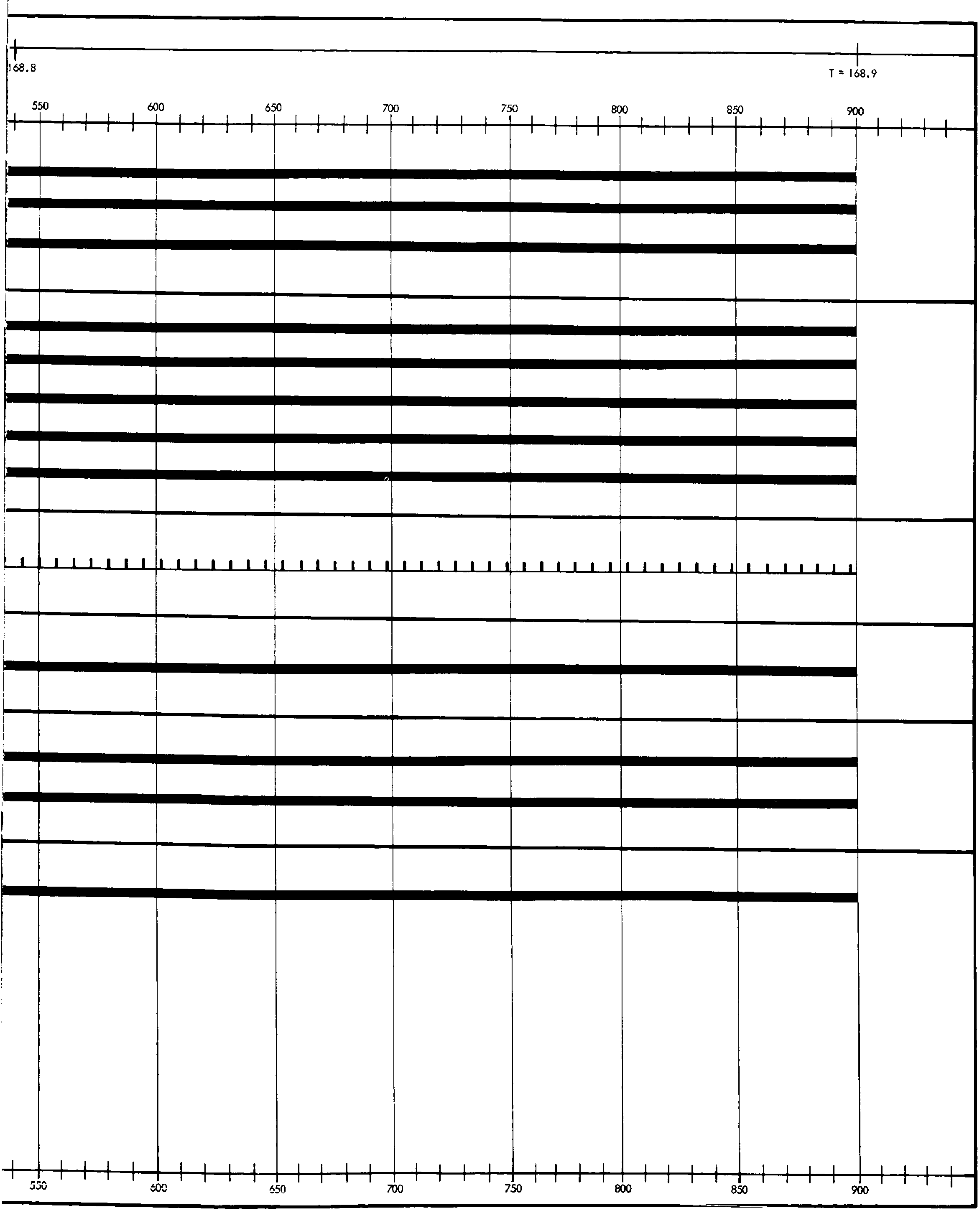


Figure 45. Mission Phase Time Line - Entry (Sheet 2 of 2)

[redacted]

PARACHUTE DESCENT PHASE

The Parachute Descent Phase begins with drogue chute deployment & ends with earth touchdown.

Figure 46 describes pictorially the sequence of operations during the Parachute Descent Phase.

Figure 47 is a two-page time-line delineation of spacecraft system activity during the Parachute Descent Phase.

CON[redacted]

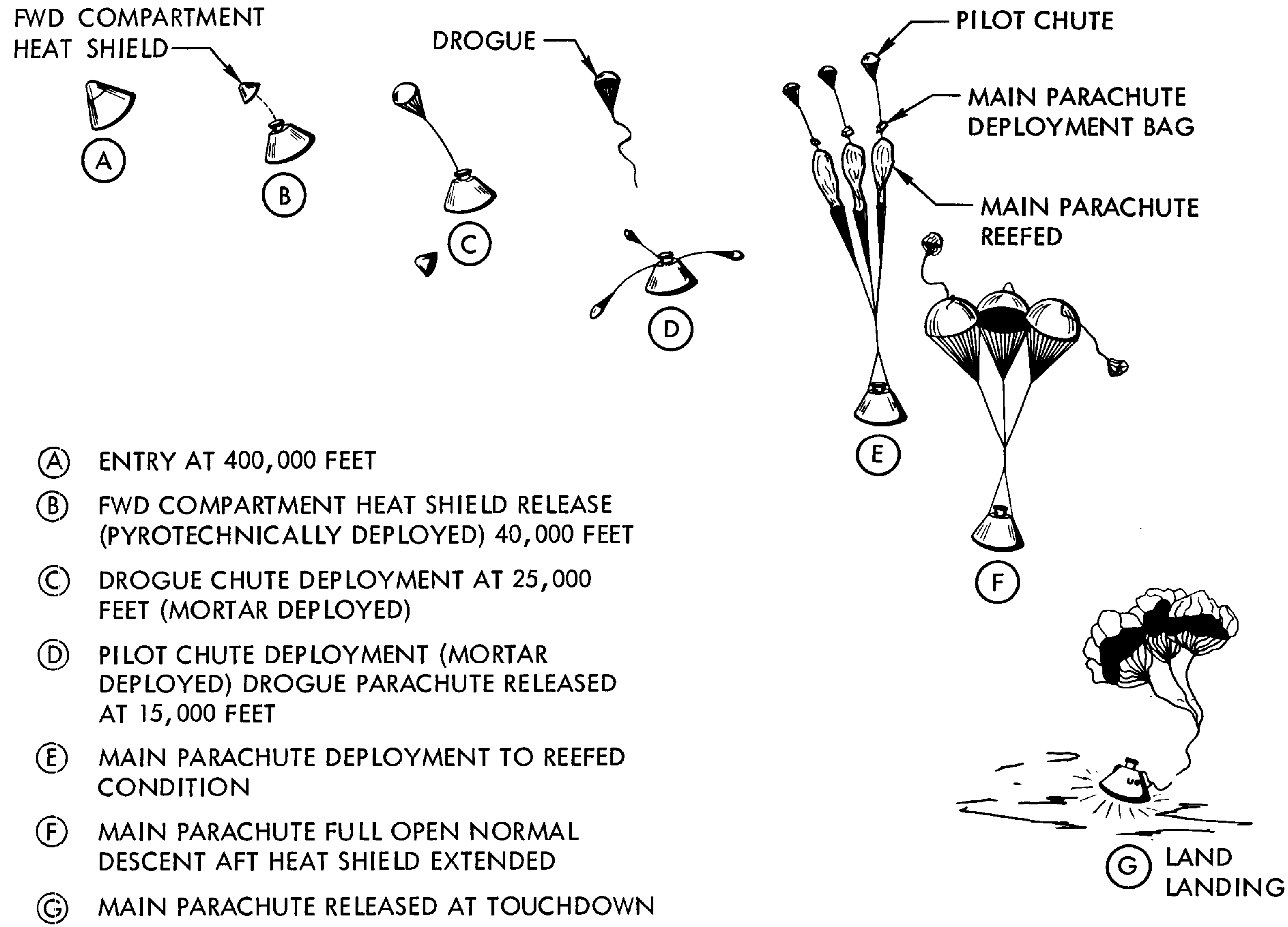


Figure 46. Parachute Descent Phase

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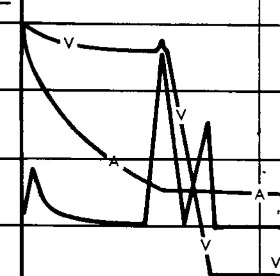
MISSION EVENTS & REQUIREMENTS	MISSION ELAPSED HOURS	EVENT DURATION	T = 168.90
DROGUE PARACHUTE DEPOLOYMENT & OPERATION	T = 168.90	30 SEC	
MAIN PARACHUTE DEPLOYMENT & OPERATION		477 SEC	
PARACHUTE MORTAR DEPLOYMENT			
PARACHUTE IN REEFED MODE			
PARACHUTE IN DISREEFED MODE			
DESCENT SEQUENCE.			
AFT HEAT SHIELD EXTENDED			
C/M STABILIZATION			
TOUCHDOWN SEQUENCE.			
MAIN CHUTES RELEASED			
	T = 169.04		

TRAJECTORY DATA - ESTIMATED

ALTITUDE 10,000 FT.	VELOCITY FPS	"G" LOAD
4	400	4
3	300	3
2	200	2
1	100	1
.7	30	
0	0	0



GOSS COVERAGE — ESTIMATED

SAN ANTONIO

POSITIONAL DATA — ESTIMATED

S/C IN SUNLIGHT

PERTINENT FUNCTIONS

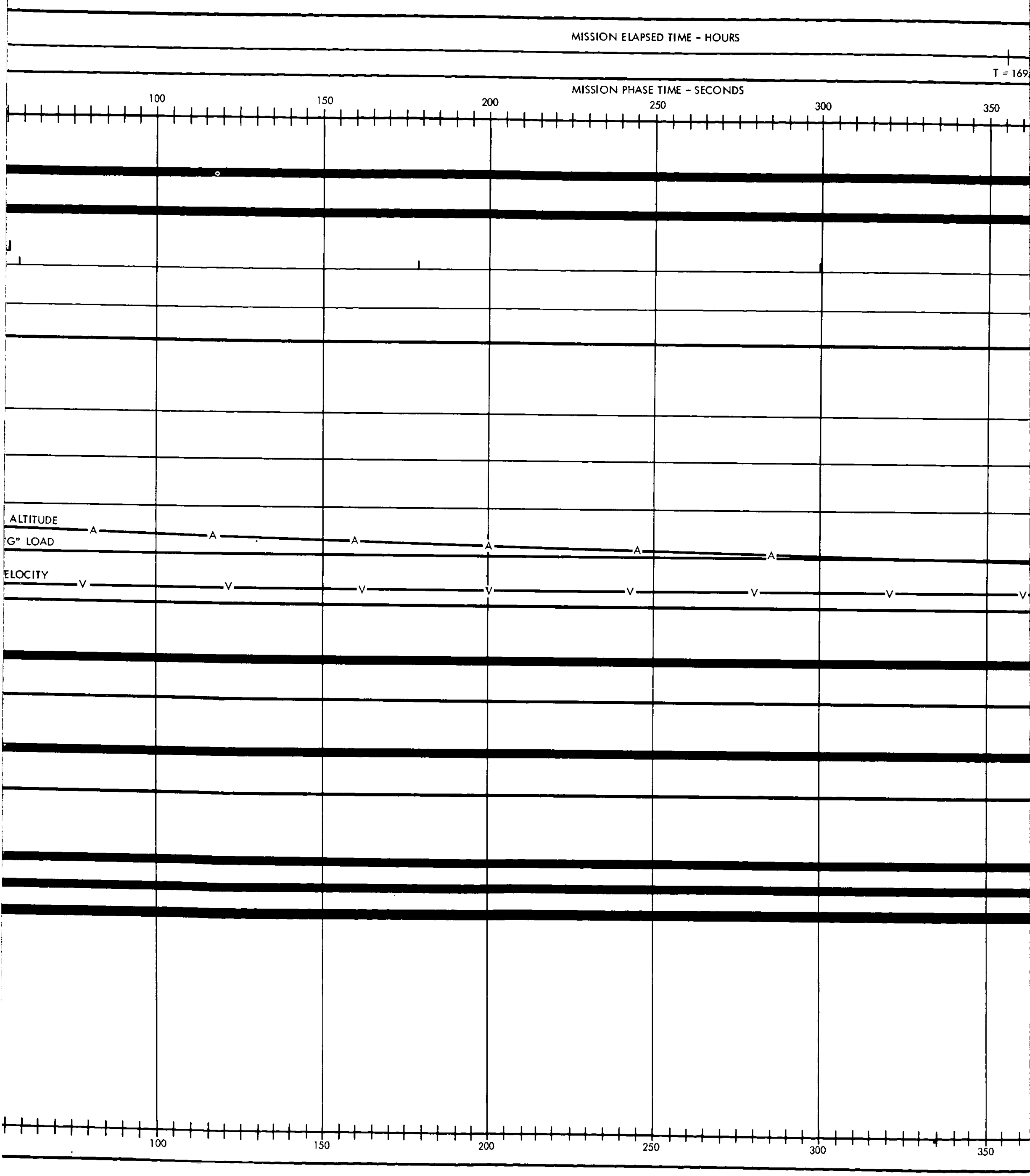
COMMUNICATIONS & INSTRUMENTATION SYSTEM

2-WAY VOICE WITH RECOVERY CRAFT

VHF RECOVERY BEACON TRANSMISSION

NEAR EARTH 2-WAY DOPPLER TRACKING OR RANGING

0 50


MISSION ELAPSED TIME - HOURS
T = 169
MISSION PHASE TIME - SECONDS
100
150
200
250
300
350
ALTITUDE
"G" LOAD
ELOCITY
100
150
200
250
300
350

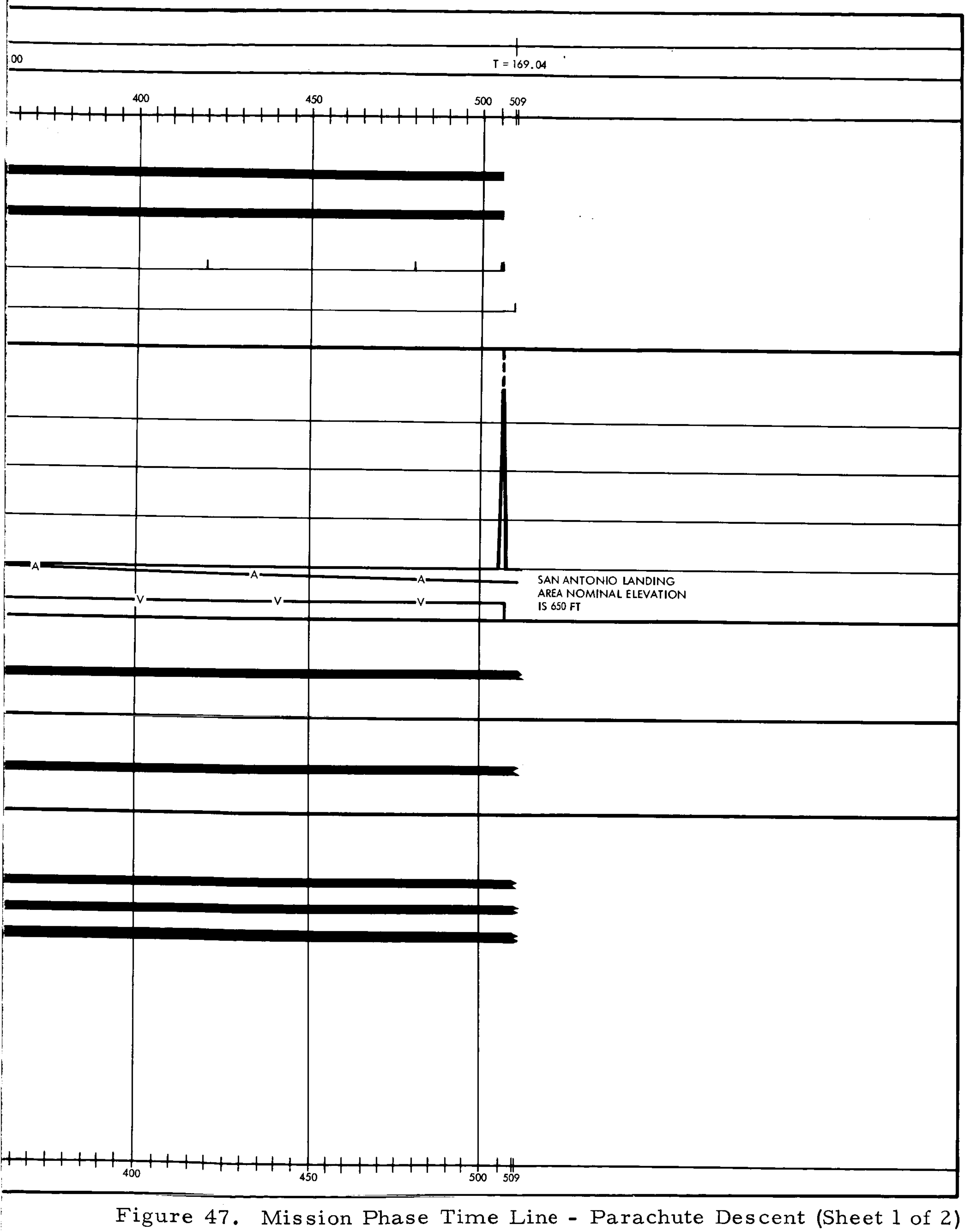
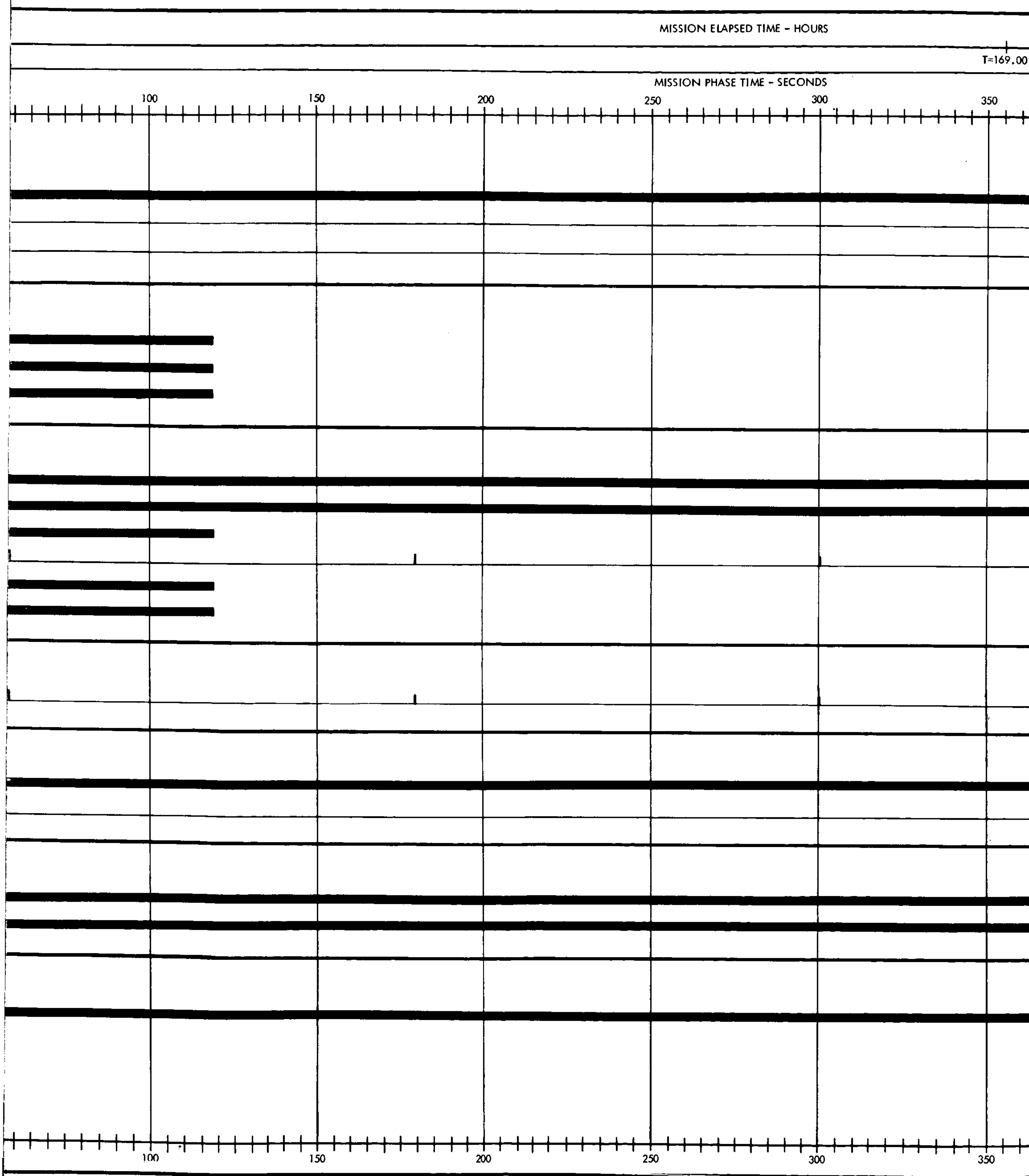


Figure 47. Mission Phase Time Line - Parachute Descent (Sheet 1 of 2)

[redacted]

PERTINENT FUNCTIONS	T= 168.00 0 5 20 50
EARTH LANDING SYSTEM	
SPACECRAFT STABILIZATION	
VELOCITY CONTROL	
IMPACT ATTENUATION	
RECOVERY AIDS	
GUIDANCE AND NAVIGATION SYSTEM	
PRIMARY INERTIAL REFERENCE	
CONTROLLED ROTATION TO SPECIFIED ATTITUDES	
G AND N ENTRY MODE	
STABILIZATION AND CONTROL SYSTEM	
SECONDARY INERTIAL REFERENCE	
ATTITUDE RATE-OF-CHANGE	
G AND N ATTITUDE HOLD MODE	
CONTROLLED ROTATION TO SPECIFIED ATTITUDE	
X-AXIS VELOCITY DATA	
TIME DATA	
C/M — REACTION CONTROL SYSTEM	
ATTITUDE IMPULSES	
ENVIRONMENTAL CONTROL SYSTEM	
PRESSURE SUIT ENVIRONMENT	
C/M VENTILATION	
CREW EQUIPMENT SYSTEM	
CREW SUPPORT & RESTRAINT	
PRESSURE SUIT ENVIRONMENT	
ELECTRICAL POWER SYSTEM	
ENTRY MAIN POWER AC & DC	
	0 5 25 50


MISSION ELAPSED TIME - HOURS
T=169.00
MISSION PHASE TIME - SECONDS
100
150
200
250
300
350

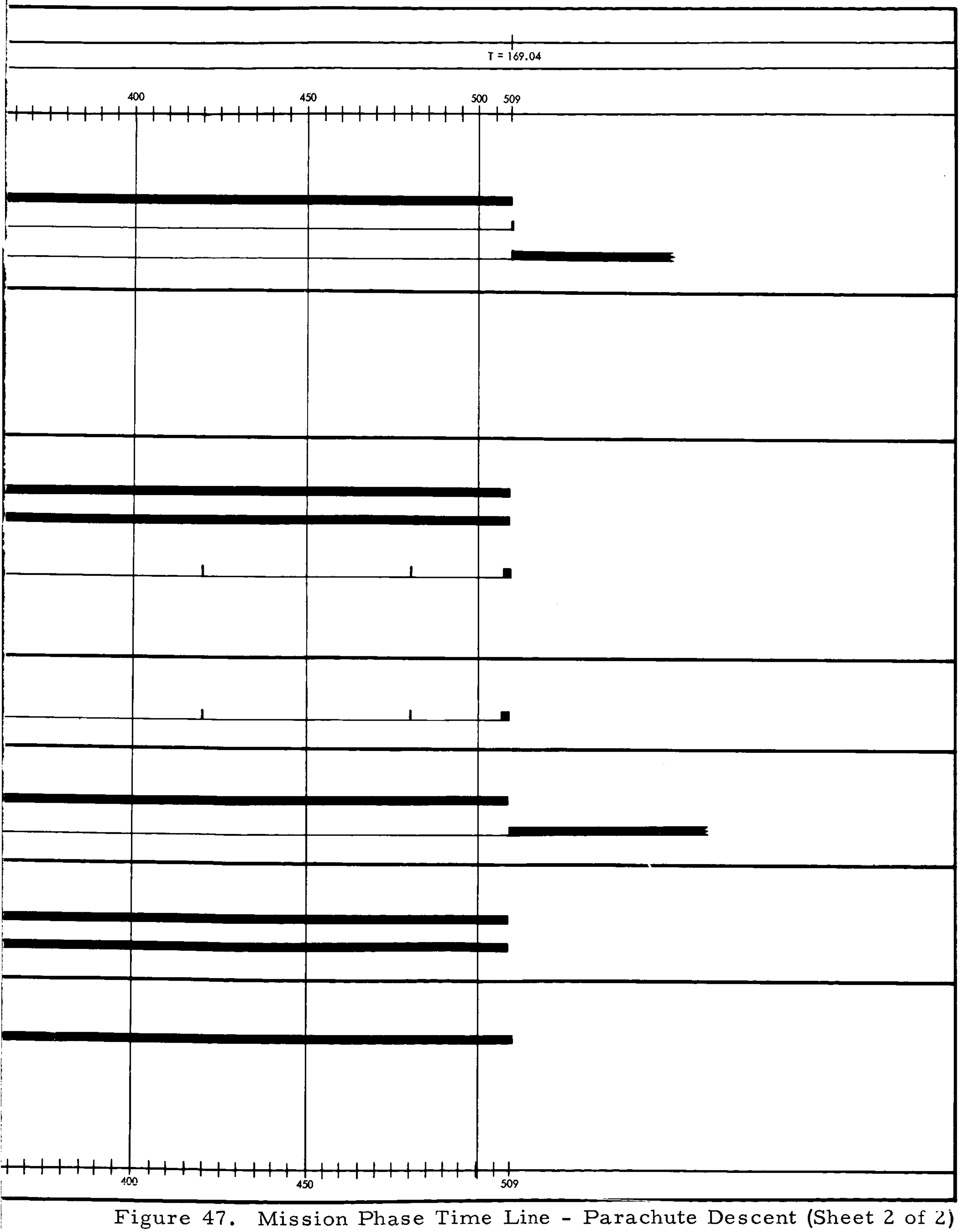


Figure 47. Mission Phase Time Line - Parachute Descent (Sheet 2 of 2)

[redacted]

APPENDIX A
SPACE VEHICLE CONFIGURATION

The Apollo Space Vehicle used in the typical lunar landing mission consists of an Apollo Spacecraft and a C-5 Launch Vehicle. Figure 48 presents overall dimensions, weight data, and relative positioning of the three C-5 booster stages and major spacecraft modules comprising the Apollo Space Vehicle.

Figure 49 presents more detailed information concerning the Launch Escape System, Command Module, Service Module, Lunar Excursion Module, and Adapter which comprise the Apollo Spacecraft. Included are dimensions, weight data, and a listing of their respective systems.

Table 2 summarizes selected performance data for the Apollo Space Vehicle at various times during the missions including weight, specific impulse and velocity.

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SPACECRAFT	
LAUNCH ESCAPE SYSTEM	
COMMAND MODULE	
SERVICE MODULE	
LUNAR EXCURSION MODULE & ADAPTER	
S-IV-B	
WEIGHT	254,662 LB
ENGINES	1 J-2
PROPULSION	200,000 LB. THRUST
FUEL	LH_2 & LO_2
S-II	
WEIGHT	1,029,024 LB
ENGINES	5 J-2
PROPULSION	1,000,000 LB. THRUST
FUEL	LH_2 & LO_2 940,000 LBS.
S-IC	
WEIGHT	4,614,341 LB
ENGINES	5 F-1
PROPULSION	7,500,000 LB THRUST
FUEL	RP-1 LO_2 4,400,000 LBS.

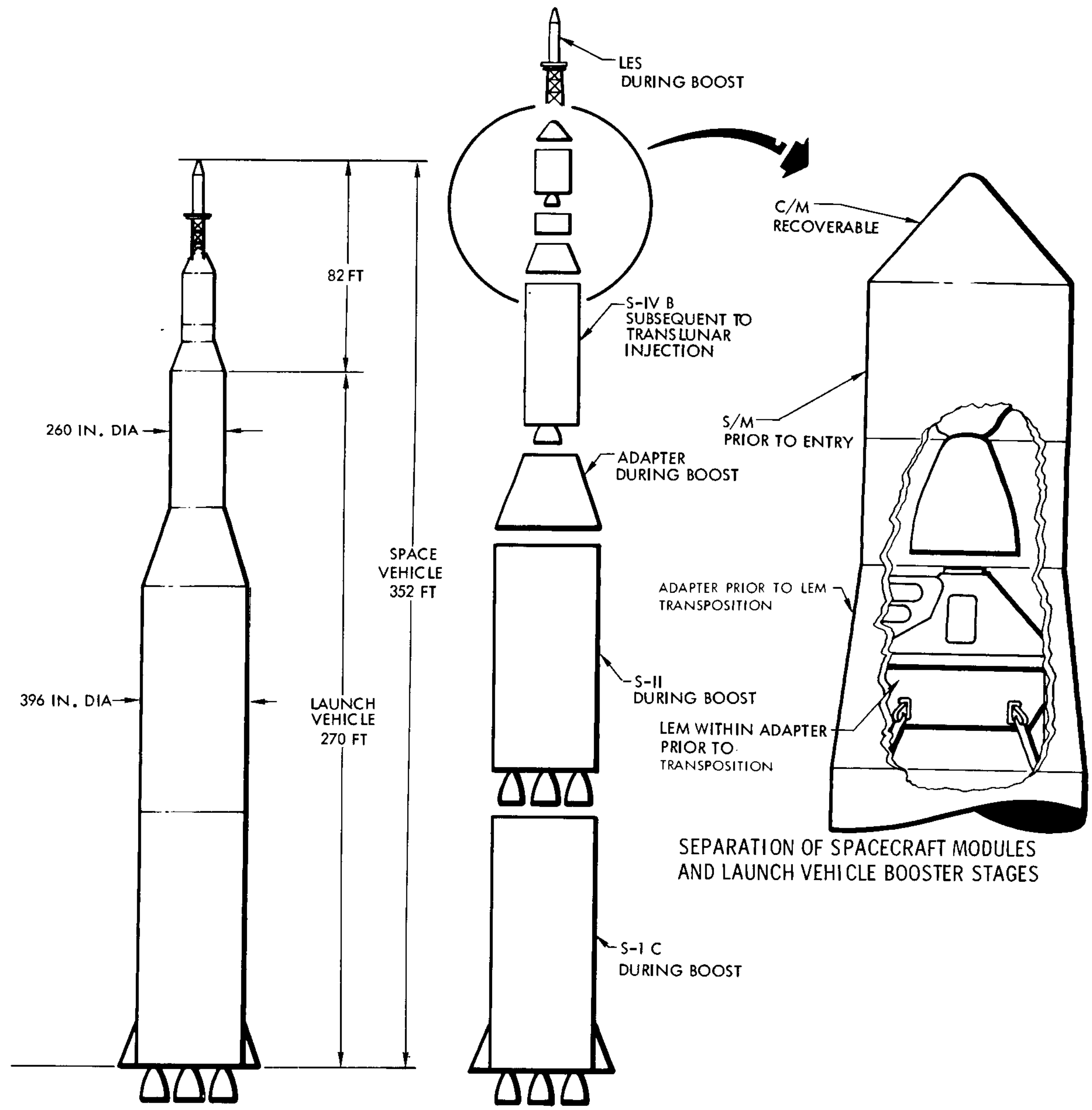

Figure 48. Apollo Space Vehicle Configuration

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LAUNCH ESCAPE SYSTEM
TAKE-OFF WEIGHT: 6600 LBS.
SYSTEMS:
NOSE CONE
TOWER ASSEMBLY
PITCH MOTOR 4,000 LB. THRUST
JETTISON MOTOR 40,000 LB. THRUST
LAUNCH EXCAPE MOTOR 120,000 LB. THRUST
BALLAST 277 LB.

COMMAND MODULE
GROSS TAKE-OFF WEIGHT: 8490 LBS.
SYSTEMS:
LAUNCH ESCAPE
STRUCTURAL AND HEAT PROTECTION
HEAT SHIELD
CREW
ENVIRONMENTAL CONTROL
IN-FLIGHT TEST
GUIDANCE & NAVIGATION
COMMUNICATIONS & INSTRUMENTATION
REACTION CONTROL
EARTH LANDING
ELECTRICAL POWER
COMMAND MODULE STABILIZATION AND CONTROL
CONTROLS & DISPLAYS

SERVICE MODULE
GROSS TAKE-OFF WT 55,848 LBS.
PROPELLANT MAX 43,050
SYSTEMS:
SERVICE PROPULSION
PROPELLANTS
OXIDIZER NITROGEN TETROXIDE N_2O_4
FUEL 50/50 UDMH & N_2H_4
REACTION CONTROL:
COMMUNICATION & INSTRUMENTATION
STRUCTURAL
ENVIRONMENTAL CONTROL
ELECTRICAL POWER
PROPELLANTS (OXIDIZER N_2O_4, FUEL NMH)
IN-FLIGHT TEST

LUNAR EXCURSION MODULE (ADAPTER)
WEIGHT (W/O ADAPTER) 24,460 lbs (UNMANNED)
ADAPTER 3,260 LBS
USABLE PROPELLANT (LEM) 15,880 LBS
SYSTEMS: (LEM)
CREW
GUIDANCE & NAV
ENVIRONMENTAL CONTROL
REACTION CONTROL
PROPULSION
COMMUNICATIONS & INST
STABILIZATION AND CONTROL
SEPARATION
IN-FLIGHT TEST
STRUCTURAL
ELECTRICAL POWER

SYSTEMS: (ADAPTER)
POSIGRADE ROCKETS
INSTRUMENTATION
ELECTRICAL
STRUCTURAL

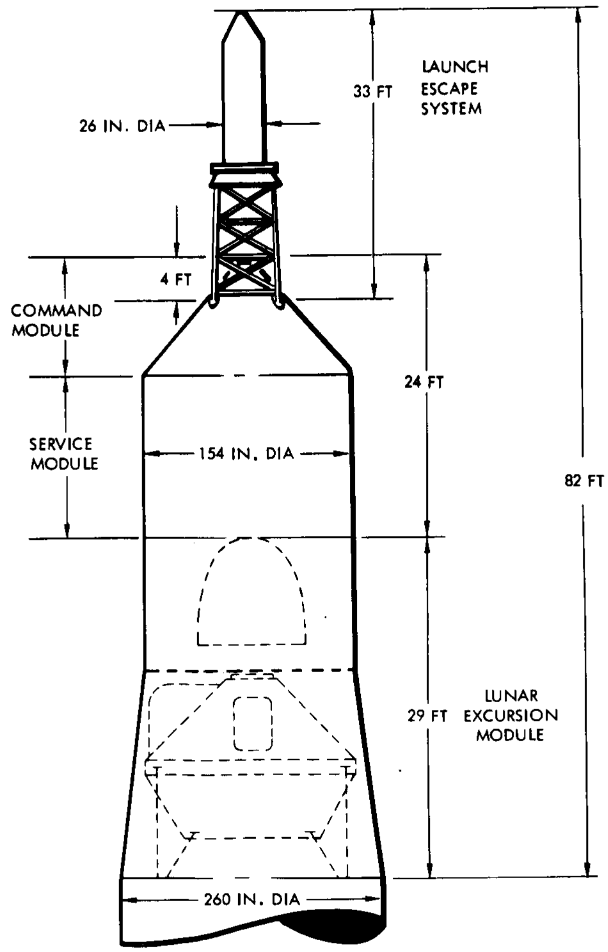


Figure 49. Apollo Spacecraft Configuration

[redacted]

TABLE 2

SPACE VEHICLE PERFORMANCE DATA

	Weight (lbs) Incremental	Weight (lbs) Balance	Velocity (FPS)
Spacecraft (C/M + S/M + LEM + LES)	198,125		
Stage 3 S-IVB	266,277		
Stage 2 S-II	1,003,800		
Stage 1 S-IC	4,631,798		
Space Vehicle at Liftoff		6,000,000	0
Less:			
S-IC Useable Propellants and Thrust Decay	4,247,698		
S-IC Stage Weight	384,100		
Vehicle	4,631,798		
Space Vehicle at S-II Ignition		1,368,202	7,615
Less:			
S-II Useable Propellants and Thrust Decay	916,000		
LES (Jettisoned 20 seconds after S-II Ignition)	6,600		
S-II Stage Weight	87,800		
Vehicle	1,010,400		
Space Vehicle at S-IVB Ignition		357,802	20,922
Less:			
S-IVB Propellant Consumed to orbit	77,271		
Weight consumed in orbit	5,000		
Vehicle	82,271		
Space Vehicle in 100 N M orbit		275,531	24,203
Less:			
S-IVB Propellant consumed to translunar injection	147,406		
SpaceVehicle		128,125	34,275
Less:			
S-IVB Stage Weight	22,000		
Interstage to payload	2,000		
Astrionics equipment	3,000		
Flight Performance Reserves	2,850		
Design reserves	4,800		
197 fps (60 M/Sec) Lunar launch window propellants	1,950		
	36,600		
Net Spacecraft Weight at Translunar Injection	91,525 lbs.		

 [redacted]

[redacted]

TABLE 2

SPACECRAFT CONFIGURATION
AT TRANSLUNAR TRAJECTORY (S/M + C/M + LEM)

	Weight (lbs) Incremental	Weight (lbs) Balance	Velocity (FPS)
S/M(Propulsion) Thrust 21,900 lbs Isp 319.5 sec.			
Spacecraft translunar injected gross weight less adapter		87,000	34,275
Less: S/M propellant consumed for translunar midcourse correction (ΔV = 300 fps)	2,429		
Spacecraft at lunar orbit injection ignition		84,571	8,336
Less: S/M propellant consumed for lunar orbit injection	22,111		
Spacecraft in lunar orbit (80 NM)		62,460	5,284
LEM 1st stage (landing) Thrust 10,000 lbs Isp 315 sec			
Lunar landing configuration: LEM			
LEM in orbit		25,000	5,284
Less: LEM propellant consumed for equip-period kick	903		
LEM in equip-period ellipse		24,097	5,671
Less: LEM propellant consumed for 50 to 1,000 ft. retro	10,627		
LEM at 1,000 ft. altitude		13,470	7.85
Less: LEM propellant consumed for surface touchdown	899		
LEM on Lunar surface		12,571	7.85
Less: LEM structure weight left on surface	2,874		0

TABLE 2

LEM 2nd Stage (Ascending) Thrust (T/W_o = .40) Isp 315 sec	Weight (lbs) Incremental	Weight (lbs) Balance	Velocity (FPS)
LEM at lunar liftoff		9,697	0
Less: LEM propellant consumed boosting to 50,000 ft orbit	4,231		
LEM in 50,000 ft. orbit		5,466	5,481
Less: LEM propellant consumed for 2-Impulse transfer to parking orbit	103		
LEM in parking orbit (manned)		5,363	5,187
Less: (2) astronauts transfer to C/M at rendezvous	540		
LEM in parking orbit (unmanned)		4,823	5,284
Transearth configuration (C/M + S/M)			
Spacecraft in parking orbit		33,571	5,284
Less: S/M propellant consumed for transearth injection	9,941		
Spacecraft at transearth injection		23,630	5,284
Less: S/M propellant consumed for transearth midcourse correction (ΔV = 300 fps)	360		
Spacecraft in transearth trajectory		22,970	8,900
Entry configuration (C/M)			
Spacecraft at entry		8,100	36,200
Spacecraft at 25,000 ft. altitude (Drogue chute deployment)		8,100	420

[redacted]

APPENDIX B

LAUNCH SITE FACILITIES

The launch facility for the C-5 space vehicle will be Complex No. 39, located at the AMR, Cape Canaveral, Florida, 28.5 degrees North latitude, and 279.5 degrees East Longitude.

The basic site plan, facilities and hardware flow plan is presented in Figure 50.

The spacecraft operations and checkout building will provide the physical equipment and space layout for receiving inspection, sub-systems performance checkout, and maintenance of the spacecraft command module, service module, adapter and LEM. The building will include a multi-story open bay area for assembly, interface systems integration, and combined systems checkout of the spacecraft modules. A fluid system test facility will provide for checkout and test of all fluid systems in the command module and service module. A vacuum chamber will be provided for reduced pressure tests of the modules.

An ordnance facility, located in a hazardous or remote area, will support space layout for inspection, modification, sub-systems performance checkout, maintenance, and bonded storage of the spacecraft solid propellant motors, explosive ordnance, pyrotechnic devices, and spare parts.

A parachute building will provide the equipment and space for receiving inspection, modification, parachute packing, and bonded storage of the command module parachute package.

A reaction control system building will provide the equipment and space for static firing tests of the reaction control system of the command module and service module. The facility will feature a high bay structure for test of assembled spacecraft. Adjacent areas will include an instrumentation control room and appropriate support shops.

A static firing facility will provide rocket propulsion test stands, propellant supply system, hydraulic fluid and gas supply system, pressurization system, instrumentation control rooms, and an emergency safety system for static firing tests of the service module propulsion system. The facility will have the capability of testing individual modules or an assembled spacecraft.

A special facility will provide space and equipment for weight and balance checkout of the command and service modules and the launch escape system (LES). Provisions will be made for physically and electrically mating the command module and the launch escape system and for subsequent weight, balance, and alignment of the mated modules.

A facility will provide radar boresight target alignment equipment to properly align the radar transponder in the mated command and service modules.

The vertical assembly building will be a high bay structure equipped for the final assembly and checkout of the integrated spacecraft and launch vehicle and will provide checkout equipment for verification of the integrated systems. The building will contain several bays for simultaneous assembly and checkout. Assembly areas will completely enclose the space vehicle during the greater part of the prelaunch checkout operations.

A launch control center will contain automatic and manual checkout and launch control equipment. The center will be linked by communications and coaxial cable with the launch pad areas and the spacecraft operations and checkout building. Final prelaunch checkout and actual launch control command of the space center will have equipment to monitor and transmit space vehicle ascent position and initial trajectory information to the mission control center at Houston, Texas.

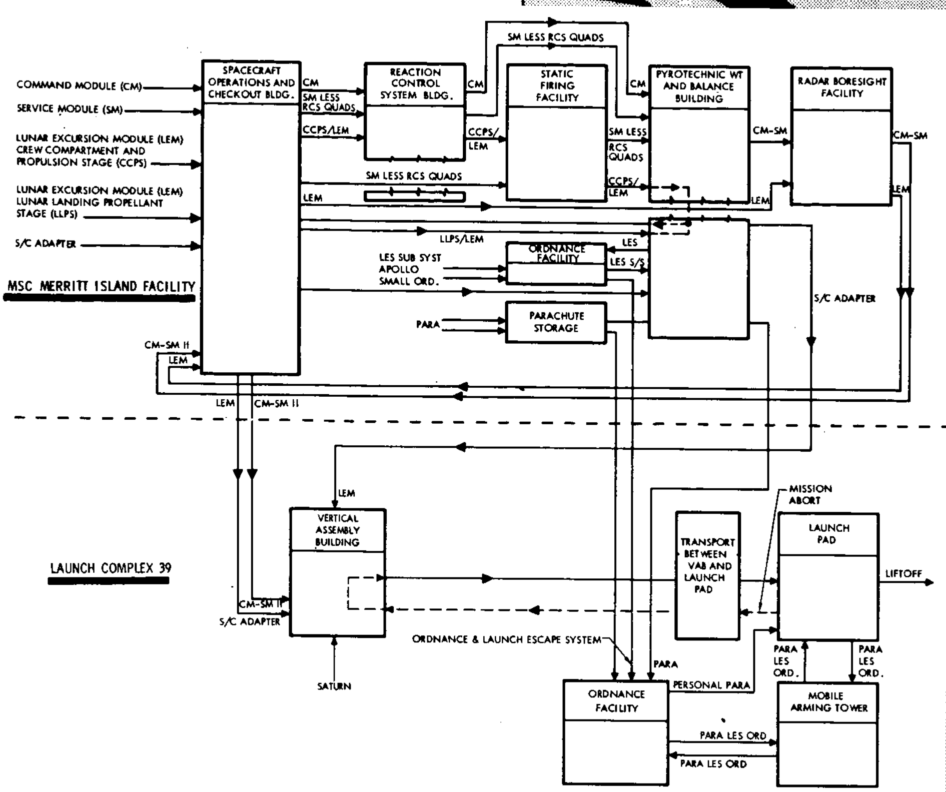
Launch pad areas will consist of large circular or rectangular concrete pads and structures equipped with a flame deflector, propellant supply facilities, electrical power, hydraulic fluid and gas supplies, decontamination facilities, coaxial and communication equipment, and personnel protective shelters. An umbilical tower will provide physical access to the space vehicle on the launch pad and will be used for crew entry to the command module.

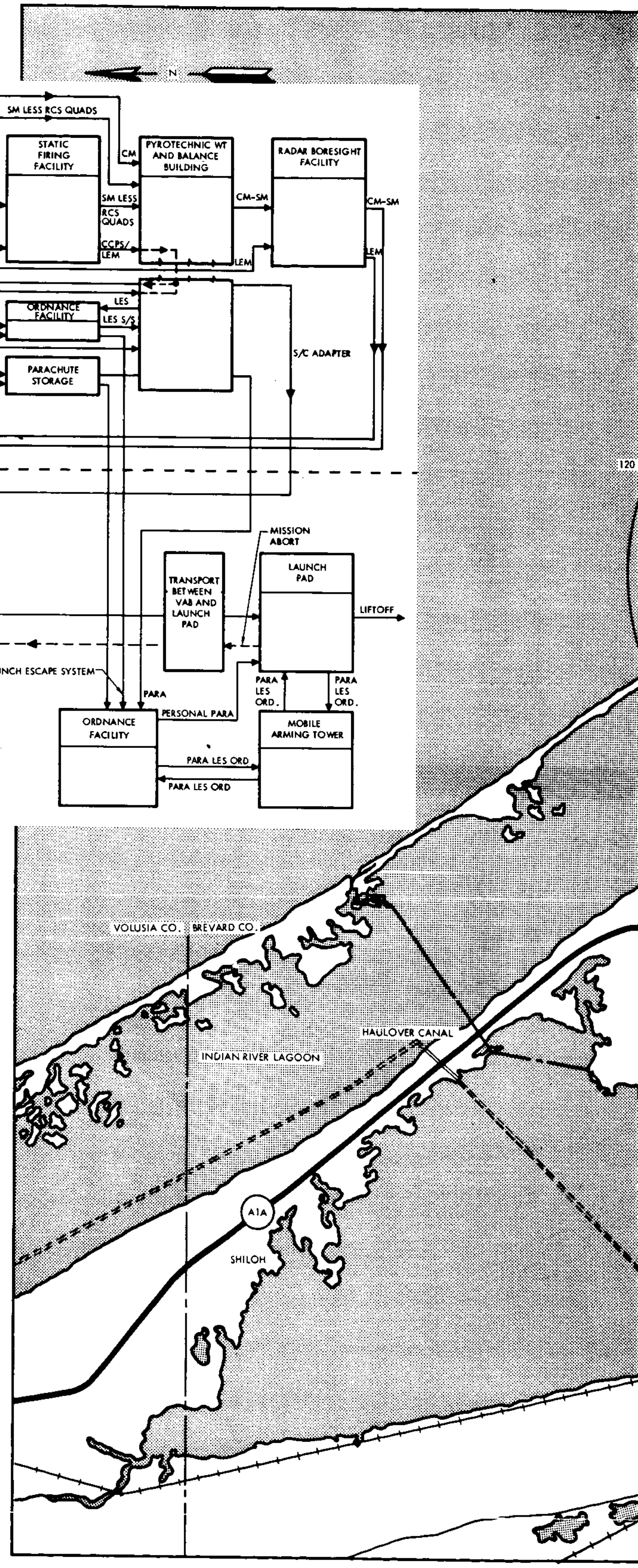
A specially constructed reinforced roadway system will interconnect all the major operational buildings and facilities of the launch complex.

The roadway will accommodate the 2,500 ton launch-transporter crawler for moving an assembled space vehicle and unbilical tower between the vertical assembly building and the launch pad area. The roadway surface will be smooth and level to minimize vibration and other adverse effects transmitted to the assembled space vehicle during transit.

A mobile arming and service tower will facilitate safe installation of flight ordnance equipment such as pyrotechnic devices, explosive units, and solid propellant motors for the launch escape system. Physical and electrical mating of the launch escape system and checkout of the command module - launch escape system intersystems will be accomplished from this service tower for all launch pads.

[redacted]





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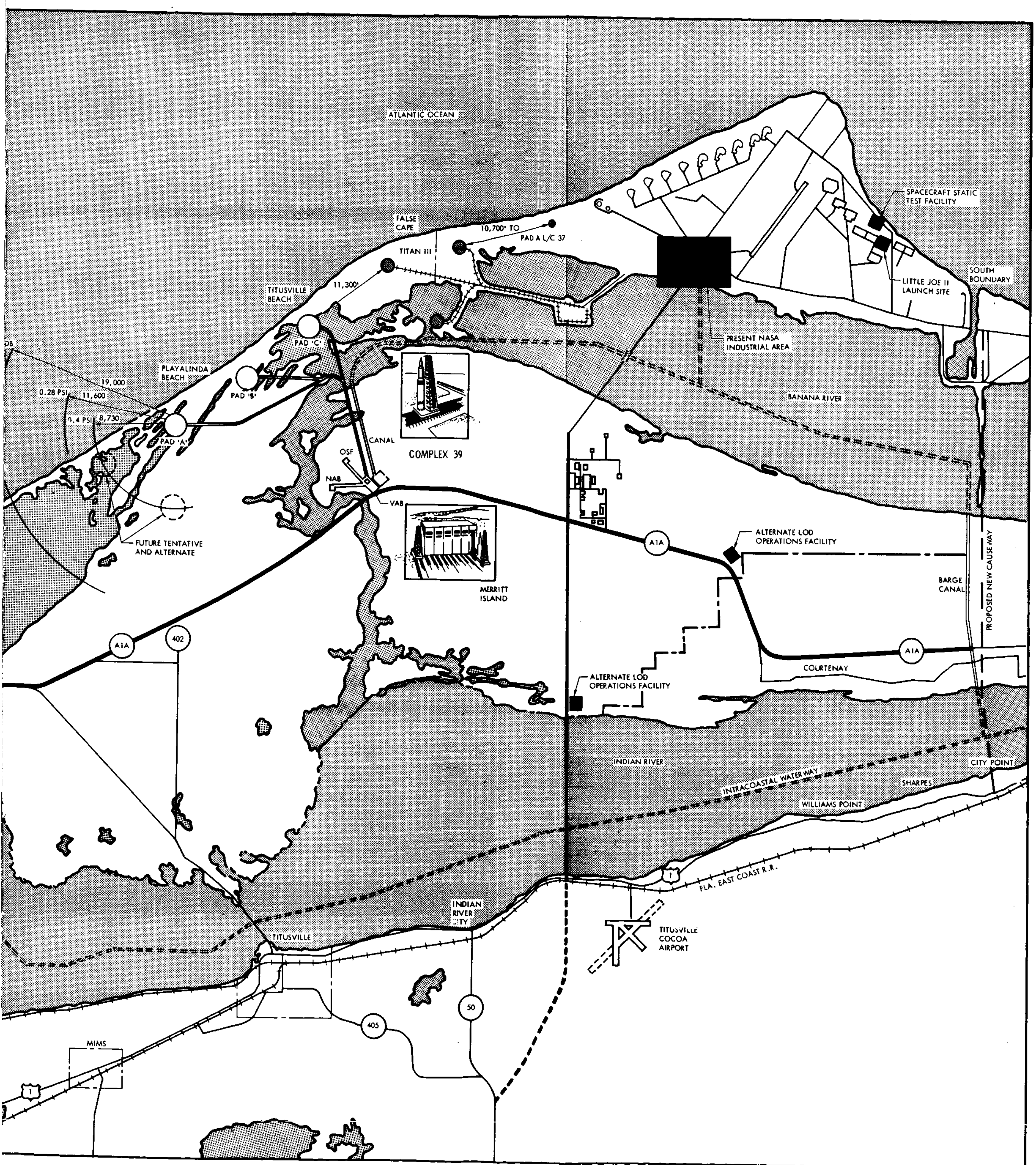

Figure 50. Cape Canaveral - Launch Site Complex

[redacted]

APPENDIX C

EARTH LANDING SITE

The landing site near San Antonio, is at 28°40' N latitude, 98°33' W longitude, located in south central Texas. Figure 51 pinpoints this location and shows a 50 mile diameter circle for scale reference. Dispersions from this impact point are not treated in this discussion.

North of San Antonio, the land slopes upward toward the Edwards Plateau, while to the South the land slopes downward towards the gulf coastal plain. The average slope is approximately 1.5 percent northwesterly. The plains region is separated from the plateau region by steep 200 to 400 foot hills and ridges. The entire area is characterized by rolling hills. The average elevation of the planned landing area is 650 feet. The surface geological structure of the plains is blackland clay and silty loam.

The planned landing area industry is predominantly agriculture, livestock, and oil.

As the landing site is located within the continental land mass of the United States, no political considerations exist.

The mean daily length of daylight during August will be 13.5 hours.

Climatological description of the landing site during the month of August is summarized below:

1. The surface winds are predominantly from the southeast at a speed of 8.1 knots.
2. Upper air winds direction are predominantly northeast at altitude of 19,300 feet and velocity of 4.0 knots.
3. Tropical storms occasionally move up from the Gulf of Mexico, producing high winds and heavy rain. Thunderstorms and heavy rains occur at all times of the year. The number of mean monthly days with thunderstormsactivity for August is four.
4. The mean monthly temperature range will be 84.2 degrees Fahrenheit. The mean maximum is 95.0 degrees while the mean minimum is 73.3 degrees. The extreme maximum is 106.0 degrees. The extreme minimum is 63.0 degrees.
5. The predominate form of precipitation during August is light rain. The mean monthly precipitation is 2.22 inches for a period of five days.
6. The mean monthly cloud cover will be 3.7 (eighths).
7. The mean monthly number of days with fog are none.

The San Antonio area, being a large population center, is well equipped with all modes of transportation facilities, and is near a large number of sizeable military establishments. Within a 60-nautical-mile radius of the landing area, there are seven military and three civilian airfields, five major rail lines, and an excellent highway network. Military airfields of particular importance are Randolph and Kelly Air Force Bases.

Because the landing area is located within the continental United States, and since there are numerous military installations in the area, the logical support of recovery forces is not expected to present problems beyond our control.

Evaluation of the combined effects of the physical environment, logistical support facilities, and the political situation, indicates that the San Antonio recovery area is a very satisfactory location for Apollo spacecraft recovery operations on land.

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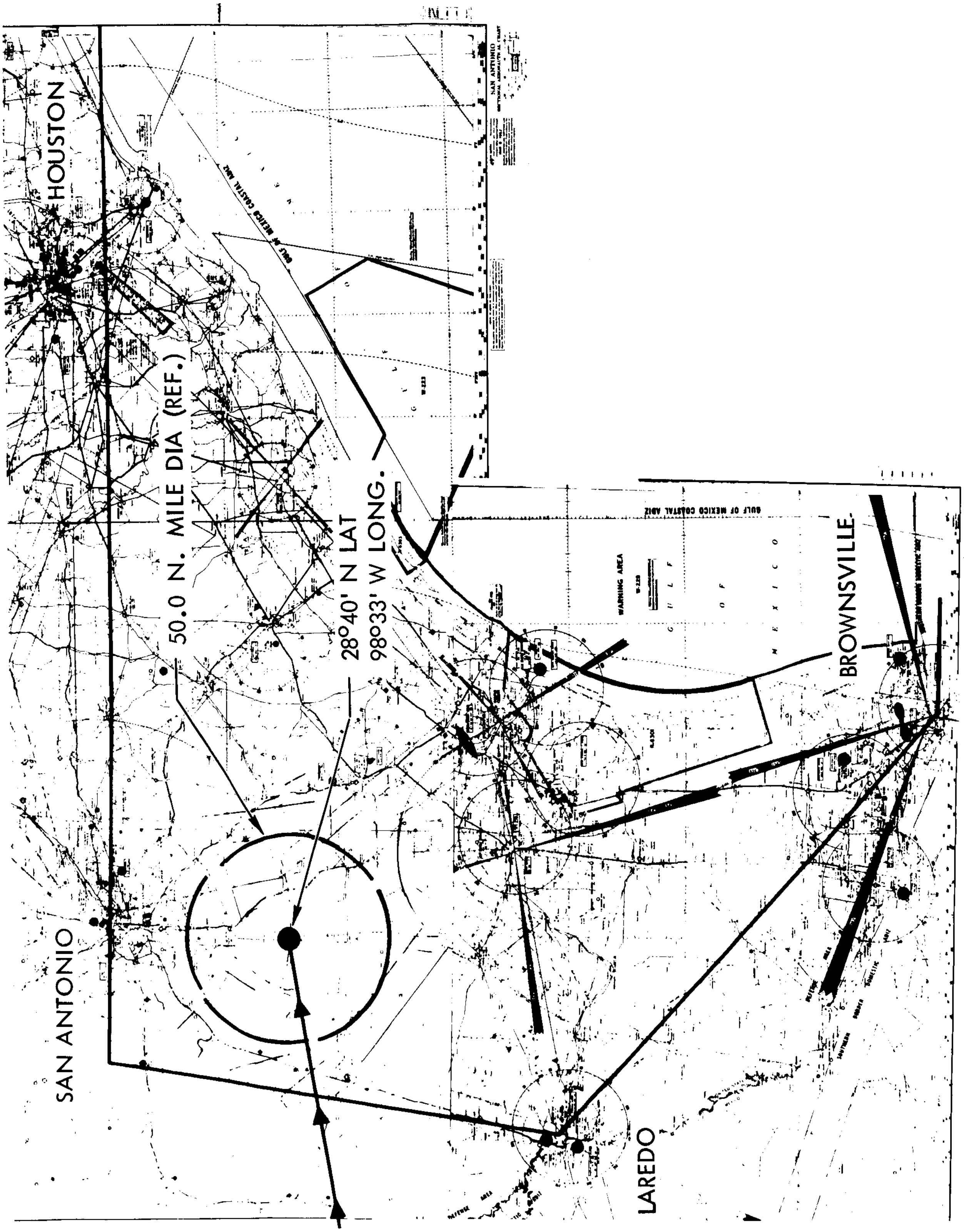

Figure 51. Earth Landing Site - San Antonio, Texas

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APPENDIX D

LIGHTING

Lighting conditions during the typical lunar landing mission are summarized in the following figures. A lighting terminator is defined as the line of demarcation between light and dark areas on the surface of a body.

Figure 52 presents lighting geometry for the dates August 14 and 17, 1967.

Figure 53 describes the relation of Cape Canaveral to the earth lighting terminator at time of launch, August 14, 1967.

Figure 54 describes the relation of the lunar landing site to the lunar lighting terminator at the time of the LEM lunar landing, August 17, 1967.

Figure 55 presents pictorially the geometric location of the earth lighting terminator during the Ascent, Earth Parking Orbit, & Translunar Injection Phases of the mission.

Figure 56 presents pictorially the geometric location of the earth lighting terminator during the Entry & Parachute Descent Phases of the mission.

[redacted]

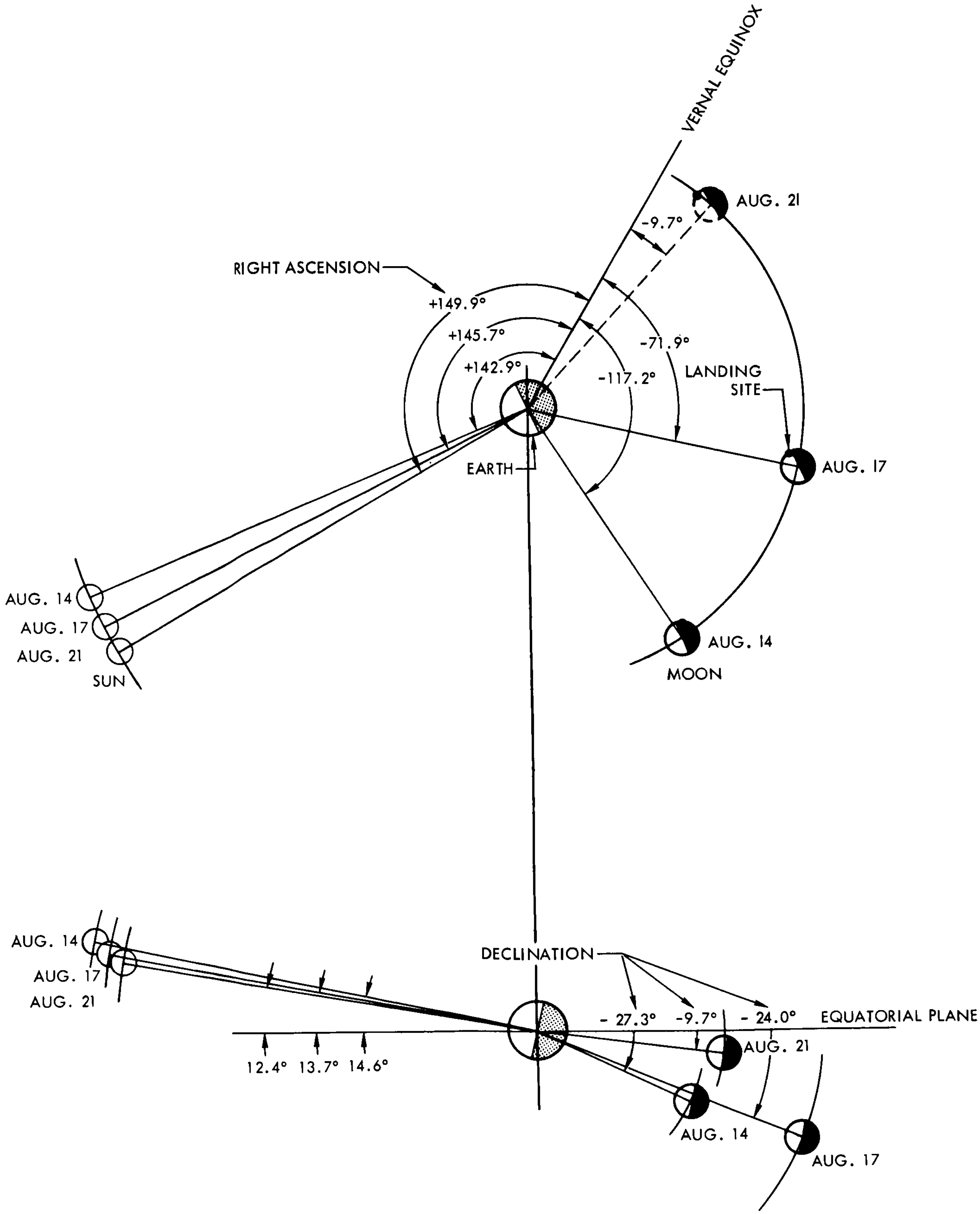

Figure 52. Lighting Terminator Geometry

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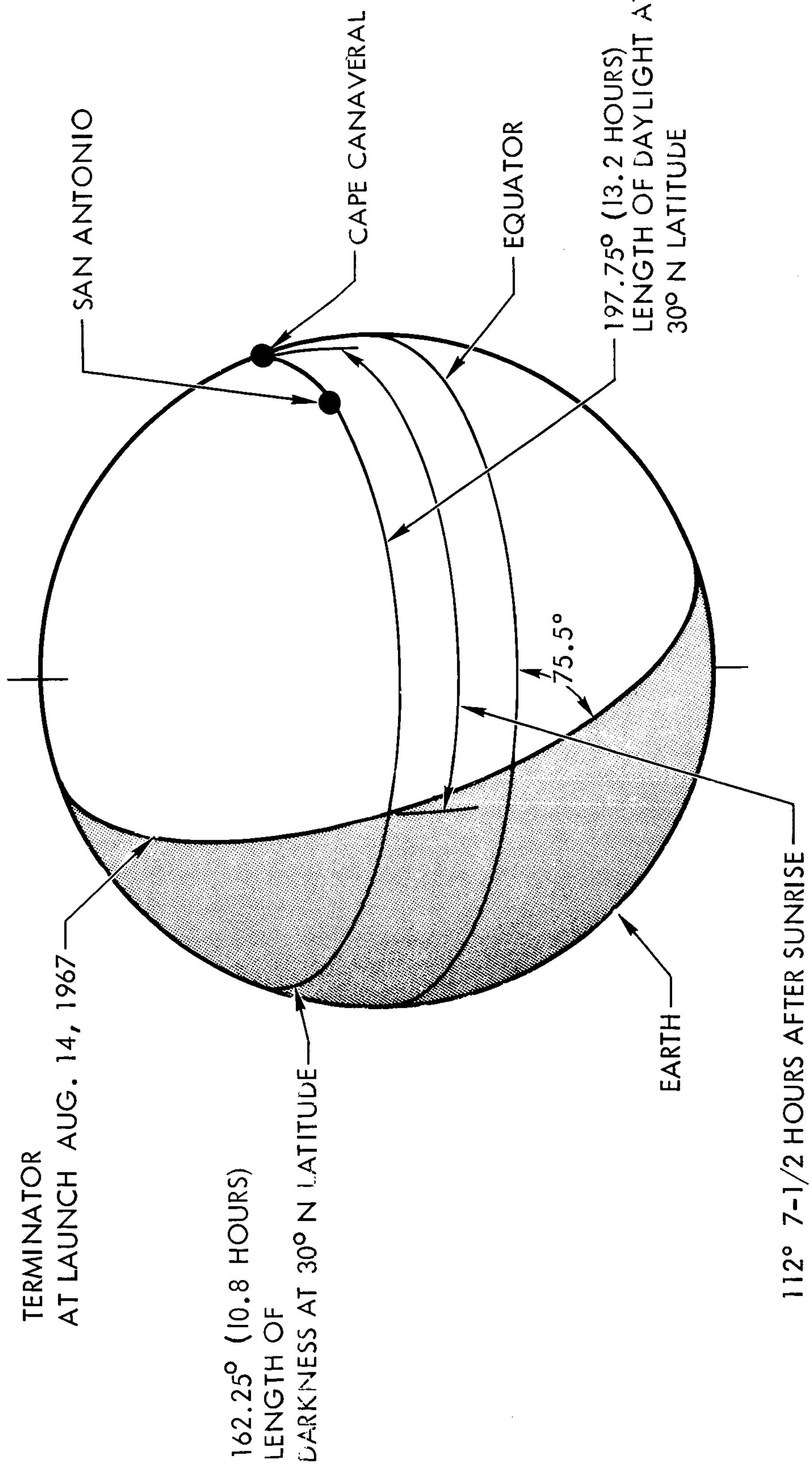

Figure 53. Earth Lighting Terminator

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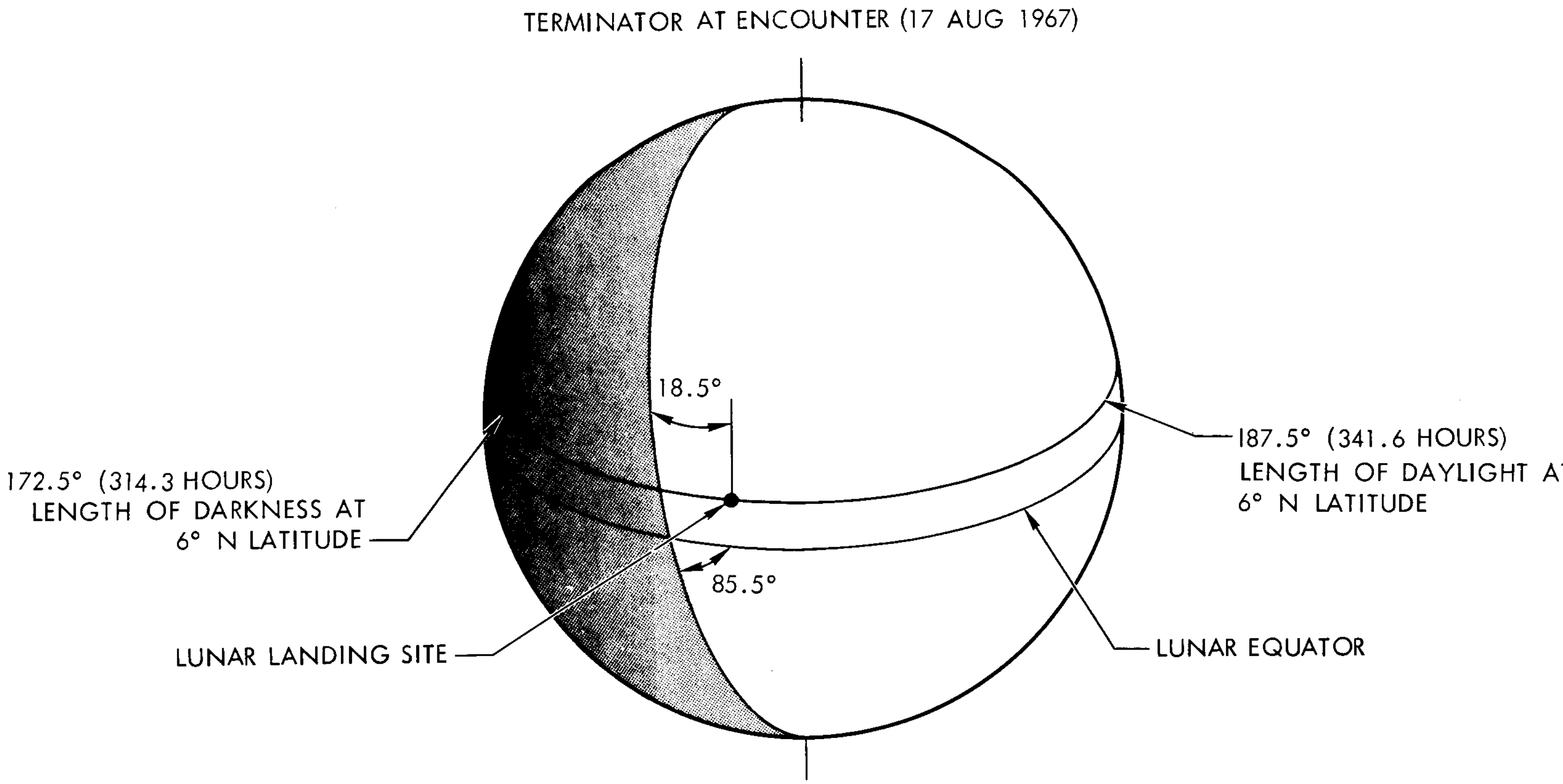

Figure 54. Lunar Lighting Terminator

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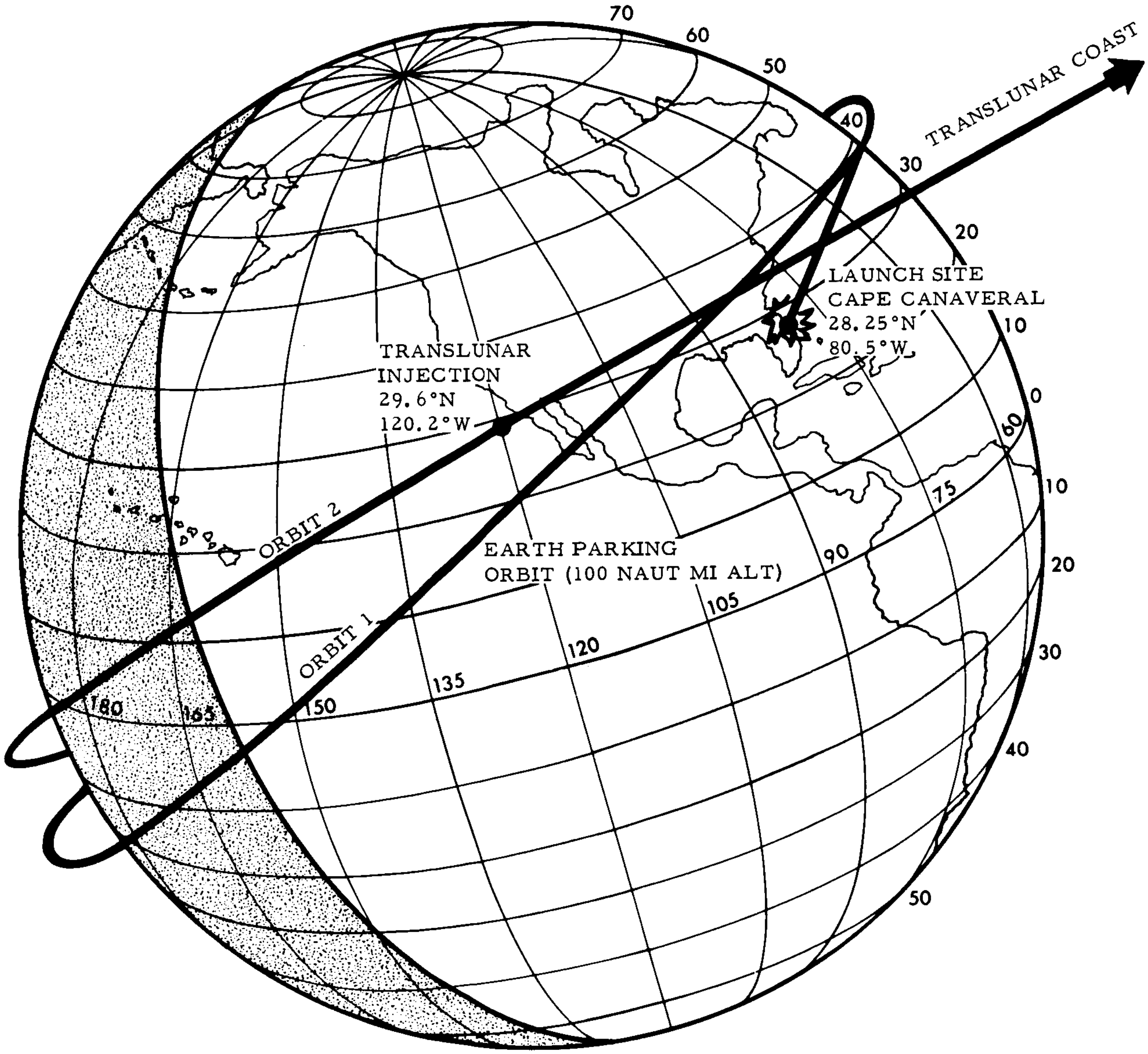

Figure 55. Earth Lighting - Ascent

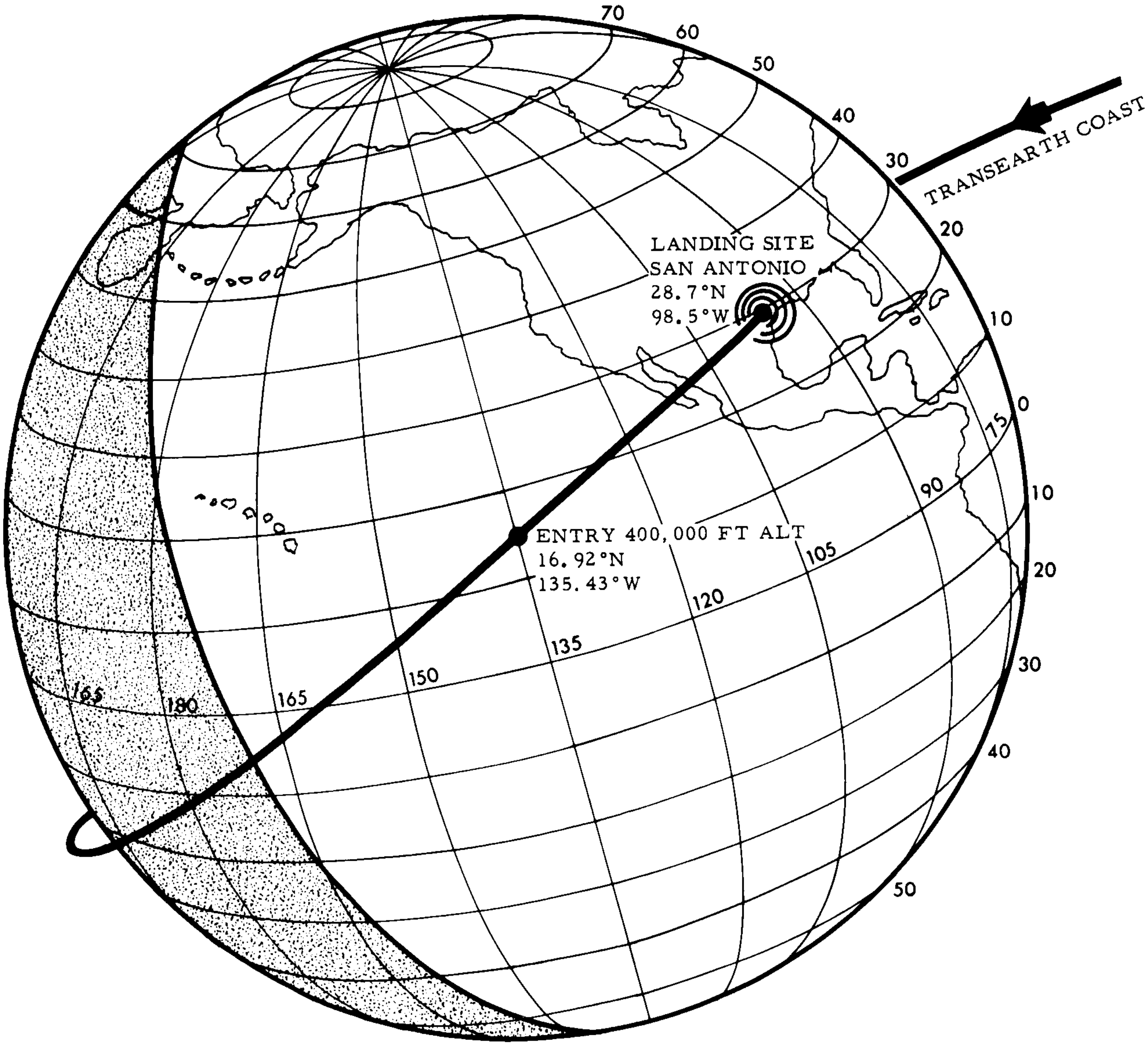

Figure 56. Earth Lighting - Entry

APPENDIX E

SPACE RADIATION

During a typical lunar landing mission, the Apollo spacecraft and crew are necessarily exposed to the hazards of space radiation. The greatest potential hazard is radiation resulting from a solar flare. Solar flares consist mainly of protons that are emitted due to intense activity on the surface of the sun. Although solar flare activity will reach a maximum during 1967-1968, events are relatively unpredictable at the present time.

Another aspect of space radiation which will affect a lunar mission is space flight through the Van Allen radiation belts. The Van Allen belts consist of charged particles which are trapped by the earth's magnetic field.

Figure 57 is a radiation belt model which indicates particle counts per second as a function of geomagnetic latitude and radial geocentric distance. Geomagnetic north has a geographical longitude of 70.1 degrees W and a latitude of 78.6 degrees N. The figure also provides a sectional view through the earth at 70.6 degrees W. The translunar coast plane of the trajectory is indicated in its relationship to radiation zones 1, 2, 3, and 4.

Figure 58 is a section through the toroidal radiation zones 1, 2, 3 and 4 in approximately the translunar flight plane. The translunar coast phase trajectory is indicated to show that its non-radial path proceeds somewhat obliquely through the annular zones.

Figure 59 presents radiation intensity and trajectory flight time as a function of distance measured in earth radii.

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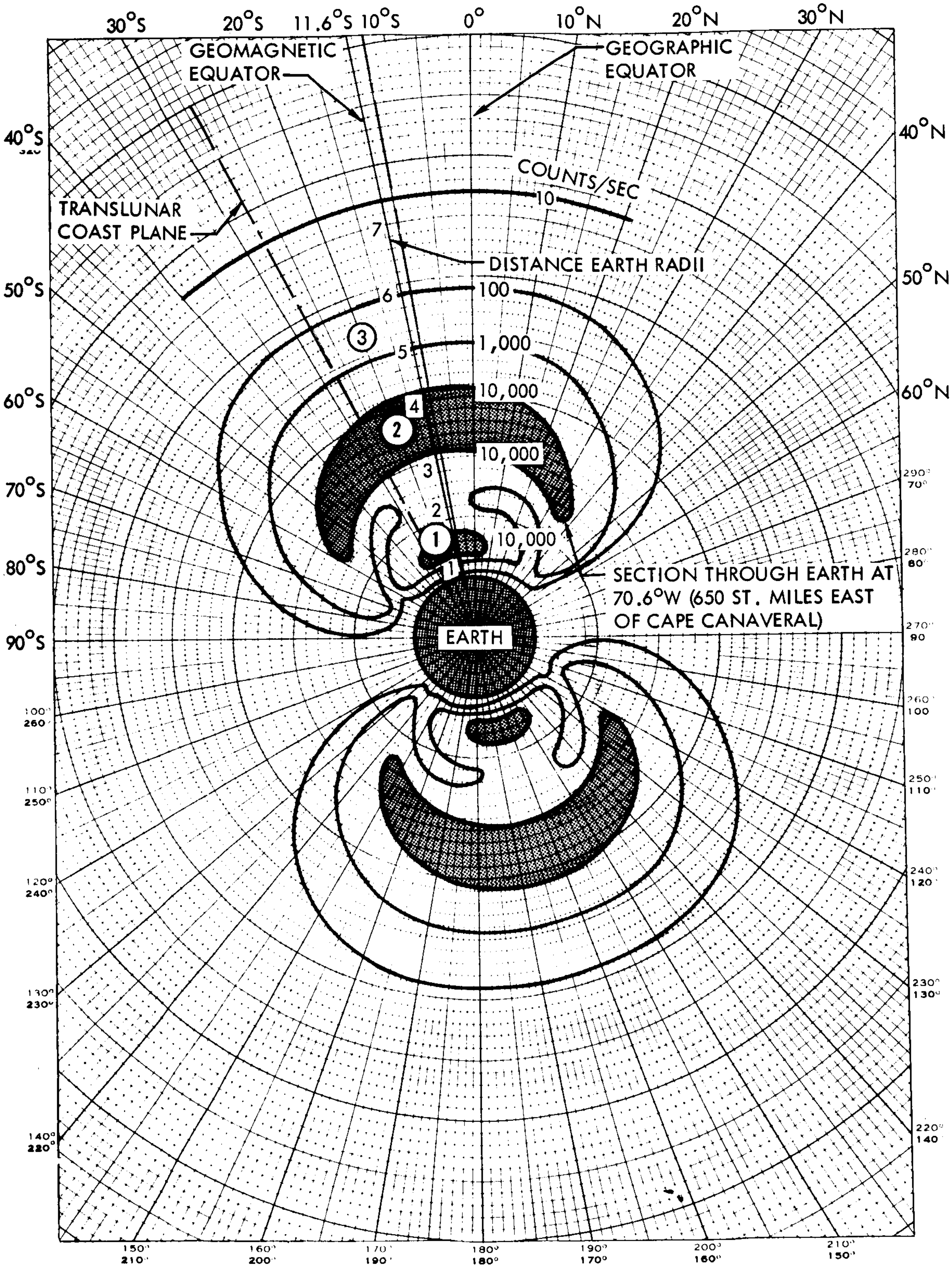

Figure 57. Van Allen Radiation Belts

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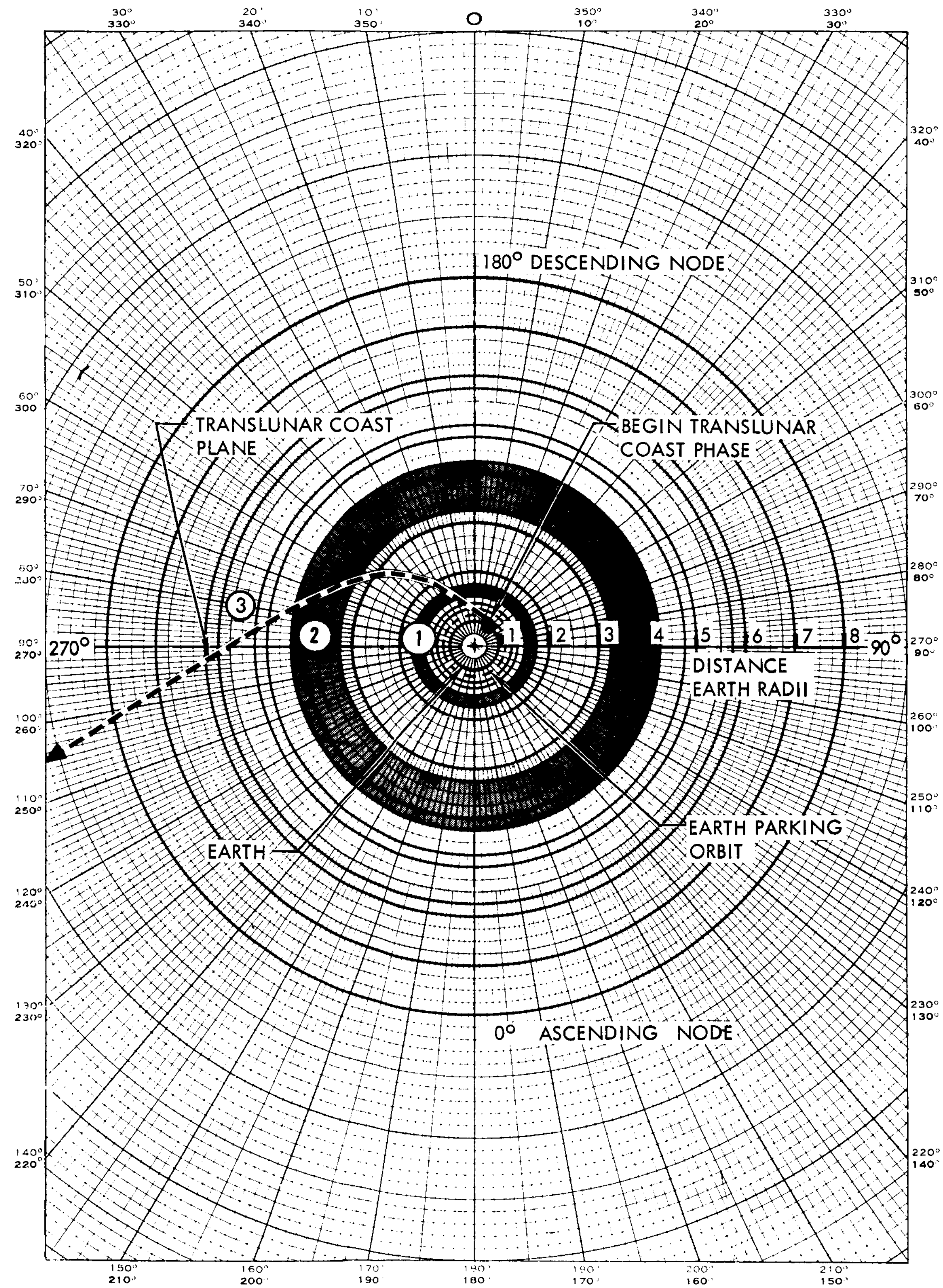

Figure 58. Mission Trajectory Geometry Thru Van Allen Radiation Belts

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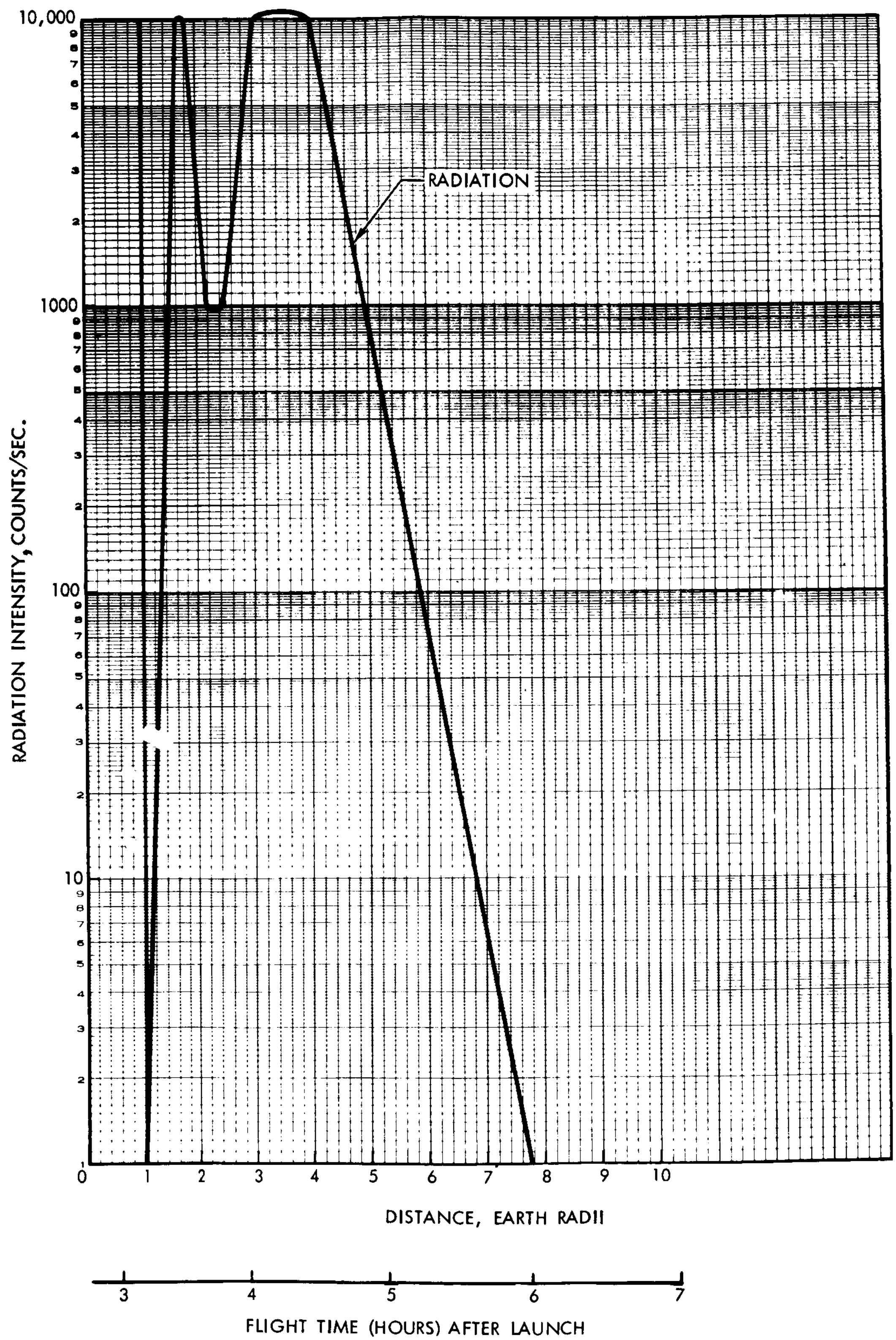

Figure 59. Radiation Intensity and Flight Time Versus Earth Radii

APPENDIX F

SPACECRAFT SYSTEMS PERTINENT FUNCTIONS

The purpose of this section is to identify and explain the pertinent functions of individual spacecraft systems. This information supplies the basic material for the time line charts of Section III of this document which delineates the spacecraft system activity during the mission.

Spacecraft systems are identified as follows:

1. Communication & Instrumentation System
2. Guidance and Navigation System
3. Stabilization and Control System
4. Service Module Reaction Control System
5. Command Module Reaction Control System
6. Service Propulsion System
7. Environmental Control System
8. Crew Equipment System
9. In-Flight Test System
10. Electrical Power System
11. Launch Escape System
12. Earth Landing System
13. Command Module Structural and Heat Protection System
14. Service Module Structural System
15. Controls & Displays System

[redacted]

For each of these systems, an introductory statement concerning the purpose of the system is followed by a listing of the major subsystems (respective components where applicable). In addition, the pertinent functions performed by each spacecraft system during the typical lunar landing mission are defined and explained together with a list of the subsystems (& components) which are required for each pertinent function.

The complete operation of each spacecraft system has been identified in terms of pertinent functions in order to facilitate the analysis of the mission operations. The occurrence of each of these pertinent functions during the mission is plotted against the mission-time-line in Section III.

Although this document presents only operations for a typical mission without malfunctions or emergencies, it will provide the basis for subsequent contingency analyses. Components and subsystems within each spacecraft system and those involved in each pertinent function are presented in a manner which should prove convenient for contingency analyses.

[redacted]

COMMUNICATIONS & INSTRUMENTATION SYSTEM

The Communication System provides transmission of voice, television, telemetry, tracking and ranging information from the spacecraft to the Earth GOSS stations. The spacecraft is also capable of receiving voice communications from the GOSS stations and processing received ranging and tracking signals for transmission back to earth. Intercommunication between crewman and communication between the Apollo Spacecraft and the LEM on the moon is also provided.

The Instrumentation System monitors spacecraft systems operations and provides appropriate displays. Provision is also made to store data on board for delayed transmission and/or for recovery within the spacecraft.

The Communications and Instrumentation System consists of the following subsystems:

RF Equipment Group

- VHF FM Transmitter
- VHR AM Transceiver
- DSIF Transponder Equipment
- C-Band Transponder
- VHF Recovery Beacon
- HF Transceiver

Antenna Equipment Group

- C-Band Antenna Equipment
- VHF Broad Band Antenna
- VHF Recovery Antenna Equipment
- Backup VHF Recovery Equipment
- HF Recovery Antenna
- 2-KMC High Gain Antenna Equipment
- 2-KMC Omni Equipment

Inter Communications Equipment Group
and Controls

- Audio Centers
- Head Sets

Data Acquisition Equipment Group

- Sensor Equipment
- Bio-Medical Equipment (NASA Supplied)
- Radiation Detection Equipment
- Scientific Instrumentation Equipment (NASA Supplied)
- Photographic Equipment
- Television Equipment

Data Handling Equipment Group

- Signal Conditioning Equipment
- Data Patch Panel Equipment
- PCM Telemetry Equipment

Data Storage Equipment Group

Spacecraft Central Timing Equipment Group

Control and Display Equipment Group

- Main Control Panel
- Antenna Control Panel
- Earth Link Control Panel
- Earth Link - Take Command Panel
- Audio Control Unit
- Television Monitor Display
- Spacecraft Central Timing Display

The space crew and the Communications & Instrumentation System perform the following pertinent functions during a normal lunar landing mission:

Near Earth Telemetry - While the spacecraft is near earth (less than 8000 miles), the C&I System telemeters sensory, radiation, and bio-medical information to GOSS. This function requires the following equipment:

- VHF FM Transmitter
- VHF Broad Band Antenna
- Sensors
- Bio-Medical Devices
- Radiation Detection Devices
- Signal Conditioners
- Data Patch Panel
- PCM Telemetry
- Central Timing Equipment

Near Earth Two-Way Voice with GOSS - While the spacecraft is near earth, the C&I System provides 2-way voice communications between the Command Module & GOSS. If the crew is not actively communicating with GOSS, the C&I System operates on a standby basis. This function requires the following equipment:

- VHF AM Transceiver
- VHF Broad Band Antenna
- Audio Centers
- Head Sets
- Antenna Multiplexer

Near Earth Two-Way Doppler Tracking/Ranging - While the spacecraft is near earth, the C&I System receives and alters signals from earth, and provides reply transmissions for use by GOSS in tracking and/or ranging. This function requires the following equipment:

- C-Band Transponder
- C-Band Antenna

Near Earth Data Storage Transmission - While the spacecraft is near earth, the C&I System transmits stored data. This function requires the following equipment:

- VHF FM Transmitter
- VHF Broad Band Antenna
- Data Storage Equipment

[redacted]

<u>C/M DSIF TV Transmission</u> - In the event of 2 KMC High Gain Antenna inoperability while in parking orbit, the C&I System provides TV operation checkout and TV transmission. This function requires the following equipment:

DSIF Transponder	TV Camera
2 KMC High Gain Antenna	Central Timing Equipment

<u>Data Storage Recording</u> - During all powered flight and flight configurations in which data transmission is impossible, the C&I System stores data for telemetry. This function requires the following equipment:

Sensors	Data Patch Panel
Bio-Medical Devices	Data Storage
Radiation Detection Devices	Central Timing Equipment
Signal Conditioners	

<u>Two-Way Voice with Belt Packs</u> - The C&I System provides 2-way voice communications between the Command Module and the Belt Packs. This function requires the following equipment:

VHF AM Transceiver	Head Sets
VHF Broad Band Antenna	Antenna Multiplexer
Audio Centers	Belt Packs

<u>Two-Way Voice with LEM</u> - The C&I System provides 2-way voice communications between the Command Module and the LEM. This function requires the following equipment:

VHF AM Transceiver	Head Sets
VHF Broad Band Antenna	Antenna Multiplexer
Audio Centers	LEM Communications

<u>DSIF Narrow Band Telemetry</u> - While the spacecraft is in deep space (greater than 8000 miles), the C&I System telemeters system, radiation, & bio-medical information to GOSS. This function requires

 [redacted]

the following equipment:

DSIF Transponder
2-KMC High Gain Antenna
Sensor Devices
Bio-Medical Devices
Radiation Detection Devices
Signal Conditions
Data Patch Panel
PCM Telemetry

<u>DSIF Two-Way Voice with GOSS</u> - The C&I System provides 2-way voice communications between the Command Module and the DSIF GOSS stations. This functions requires the following equipment:

DSIF Transponder
2-KMC High Gain Antenna
Audio Centers
Head Sets

<u>DSIF TV Transmission</u> - The C&I System provides TV transmission from the Command Module to the DSIF GOSS stations. This function requires the following equipment:

DSIF Transponder
DSIF Transponder Amplifier
2-KMC High Gain Antenna
TV Camera

<u>DSIF Doppler 2-Way Tracking/Ranging</u> - The C&I System provides 2-way Doppler tracking of the spacecraft by the DSIF GOSS stations. Ranging by GOSS is accomplished by the C/M equipment usage configuration. This function requires the following equipment:

DSIF Transponder
2-KMC High Gain Antenna

<u>DSIF Two-Way Voice Relay to GOSS</u> - The C&I System provides 2-way voice relay from the belt packs at a remote location (lunar surface) or the LEM to GOSS via the Command Module. This function requires the following equipment:

VHF AM Transceiver
DSIF Transponder
VHF Broad Band Antenna
2-KMC High Gain Antenna
Antenna Multiplexer
Audio Center

[redacted]

DSIF Broad Band Telemetry - While the spacecraft is in deep space, the C&I System telemeters ~~system~~, radiation, and bio-medical information to GOSS. Broad band telemetry has a greater information capacity than does narrow band. This function requires the following equipment:

- DSIF Transponder
- DSIF Transponder Amplifier
- 2-KMC High Gain Antenna
- Sensors
- Bio-Medical Devices
- Radiation Detection Devices
- Signal Conditioners
- Data Patch Panel
- PCM Telemetry Equipment
- Central Timer

DSIF Data Storage Transmission - While the spacecraft is in deep space, the C&I System transmits stored data to GOSS. This function requires the following equipment:

- DSIF Transponder
- DSIF Transponder Amplifier
- 2-KMC High Gain Antenna
- Data Storage Equipment
- Central Timer

Two-Way Voice with Recovery Craft - Prior to earth impact of the Command Module, the C&I System provides 2-way voice communications between the Command Module and the recovery craft. This function requires the following equipment:

- VHF AM Transceiver
- VHF Recovery Antenna
- Audio Centers
- Head Sets
- Antenna Multiplexer

VHF Recovery Beacon Transmission - Prior to earth impact of the Command Module, the C&I System provides a direction finding beacon to aid recovery craft in finding the C/M. This function requires the following equipment:

- VHF Recovery Beacon
- VHF Recovery Antenna
- Antenna Multiplexer

 [redacted]

[redacted]

GUIDANCE AND NAVIGATION SYSTEM

The Apollo Guidance and Navigation System is a semi-automatic spacecraft guidance and navigation system that is directed and operated by the space crew to provide the G and N display and control signals required by the space crew, the Stabilization and Control System, the Service Module Propulsion System, the Service Module Reaction Control System, and the Command Module Reaction Control System.

During the initial phases of a lunar landing mission, when the S-IVB is a part of the spacecraft, the Apollo Guidance and Navigation System will only generate the G and N monitoring signals that will be displayed to the space crew by the Stabilisation and Control System. After Translunar Injection and after the S-IVB stage has been separated from the spacecraft the Apollo Guidance and Navigation System will then perform all of the G and N functions and generate all of the G and N signals that are required by the spacecraft to complete the lunar landing mission.

The Apollo Guidance and Navigation System will be designed by MIT and will be delivered to S&ID as NASA furnished equipment. The Guidance and Navigation System will include the following major components:

Inertial Measurement Unit (IMU)

Manual Gyro Torquing Controls

Apollo Guidance Computer (AGC)

AGC Controls

AGC Displays

[redacted]

Coupling Display Unit (CDU)

CDU Manual Controls

CDU Displays

Power and Servo Amplifier (PSA)

PSA Displays

Scanning Telescope (SCT)

Optical Hand Controller

SCT Mechanical Hand Controller

SCT Displays

Sextant (SXT)

Optical Hand Controller

Optical Star Tracker

SXT Displays

The space crew and the Guidance and Navigation System performs the following pertinent functions on a normal lunar landing mission:

Primary Inertial Reference - A primary inertial reference (attitude and acceleration) will be established and maintained by the G and N System. The mode of the reference framework may be selected with respect to a number of coordinate axes including those of the earth and the moon. The G and N inertial reference may be shutdown, when it is not needed on the mission, and new inertial references may be established during the mission by fine aligning the IMU stable element.

The G and N components that are normally required by the maintenance of this primary inertial reference are the IMU, PSA, AGC, CDU, and their respective normal controls & displays.

The establishment of a primary inertial reference, on the launch pad prior to take-off, will require the G and N components listed above plus special aerospace support equipment on the ground.

The establishment of a primary inertial reference during the mission will include an IMU fine alignment procedure and this will require the G and N components listed above plus the use of the SCT and SXT with their normal controls and displays.

<u>SCS Monitor Mode</u> - In the SCS monitor mode the G and N system will maintain a primary inertial reference and will generate attitude displacement signals when the spacecraft attitude deviates from the attitude reference. Attitude error signals will also be generated when the measured angles of the IMU deviate from the commanded angles of the AGC. Attitude displacement and error data will be displayed to the space crew by the FDAI of the Stabilisation and Control System. No control signals will be transmitted to the attitude or thrust control systems in this moniter mode.

The G and N system may also generate acceleration and velocity data during this mode and acceleration monitor data may be displayed by the SCS entry display panel during take-off, the boost phases and the translunar injection.

A prerequisite of this function is the establishement and maintenance of a primary inertial reference. The G and N components that would then normally be required by this function are the IMU, PSA, AGC, CDU, and their respective controls & displays.

Earth Parking Orbit and Ephemerides - A procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the present parking orbit, obtain a GOSS determination of the parking orbit, compare the on-board data with the GOSS determined data, and then determine the final corrected values of the present parking orbit.

The on-board procedure will be to have the S-IVB booster stage orient and stabilize the spacecraft as required, perform a series of SCT and SXT navigational sightings on known landmarks and starts, insert this sighting data into the AGC, initiate a computer program that will first determine the moving positions and velocity values of the parking orbit and will then predict the ephemerides of the future parking orbits.

The on-board computations and navigational sightings assume that the spacecraft AGC will have been provided with four body equations of motion that define a reference parking orbit and also defines what the measured angles should be between known landmarks and stars if the spacecraft were at a given point on the reference parking orbit. The spacecraft computer (AGC) will then accept actual measured angle data and on the basis of the difference between the reference angle and the actual measured angle, the computer will be able to calculate the actual degree of deviation that exists from the reference parking orbit.

A prerequisite of this function is the establishment and maintenance of a primary inertial reference. All major G and N components and their normal controls and displays would then be required to perform this function.

<u>Translunar Injection Parameters</u> - The procedural description of this system function is as follows: The general procedure will be to perform an on-board determination of the desired translunar injection parameters, obtain a GOSS determination of the injection parameters, compare the on-board data with the GOSS determined data and then determine the final corrected values of the translunar injection point and program.

The on-board procedure assumes that the spacecraft computer will have been provided with four body equations of motion that define a reference earth parking orbit, a reference translunar injection point and program, and a reference translunar trajectory that passes through a specified key lunar aiming point. The spacecraft computer will then accept actual parking orbit data and on the basis of the difference between the actual and the reference parking orbit the computer will then be able to calculate the desired translunar injection point and program that is required to hit the specified key lunar aiming point.

Only the AGC with its normal controls and displays will be required to perform this system function.

<u>G and N Attitude Hold Mode</u> - Attitude hold will be the normal spacecraft function of this mode. An adjustable deadband for the different functional requirements of attitude hold is available and may be selected by channel on the SCS control panel. Attitude displacement (control and display) signals will be generated by the IMU to drive the ball attitude indicator of the FDAI and control the spacecraft attitude by directing the appropriate Reaction Control System. Attitude error display signals will

be generated by the CDU for each axis channel and will show the difference between the IMU gimbal position and the AGC commanded position. Attitude error signals will be displayed to the crew on the SCS FDAI.

A prerequisite of this function is the establishment and maintenance of a primary inertial reference. The G and N components that would normally be required by this function are the IMU, PSA, AGC, CDU, and their respective controls & displays.

Controlled Rotation to Specified Attitudes - This function will consist of the automatically controlled attitude maneuvers that are directed by the G and N System during a G and N Attitude Hold Mode.

G and N automatically controlled attitude maneuvers will be accomplished in two ways during a G and N Attitude Hold Mode. The first method involves computer program inputs of specific commands that are inserted through the computer keyboard to direct the AGC. The second method involves special enabling commands to the AGC directing the computer to generate signals that cause the spacecraft to maneuver to specified attitudes.

Manual methods supplied by the SCS, can also be used to override or interrupt the G and N Attitude Hold Mode and direct the appropriate Reaction Control System to rotate the spacecraft to specified attitudes.

Prerequisites of this automatic G and N maneuver function are the establishment and maintenance of a primary inertial reference, and the performance of a G and N Attitude Hold Mode. The G and N components that would then normally be required by this function are the IMU, PSA, AGU, CDU, and their respective controls and displays.

<u>Present Translunar Trajectory</u> - The procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the present translunar trajectory, obtain a GOSS determination of the trajectory, compare the on-board data with the GOSS determined data and then determine the final corrected values of the trajectory.

The on-board G and N procedure will be to perform an attitude hold procedure, generate signals to maneuver the spacecraft to specified attitudes, perform a series of SCT and SXT navigational sightings on known landmarks and starts, insert the navigational sighting data into the AGC, initiate computer programs that will first smooth and average out the various navigational sightings and will then compute the present translunar trajectory.

Navigational sightings will consist of obtaining directional cosines of known landmarks and stars. The translunar navigational sightings may be scheduled at approximately half hour intervals, a series of approximately 10 navigational sightings may be averaged together to produce each on-board present trajectory determination and each of the 10 navigational sightings will include a number of SCT and SXT data inputs into the AGC.

The on-board G and N computations and navigational sightings assume that the spacecraft computer will have been provided with four body equations of motion that define a reference translunar trajectory and also defined what the measured angles or directional cosines should be between known landmarks and stars if the spacecraft were at a given point on the reference trajectory. The spacecraft computer will then accept actual measured angle data and on the basis of the difference between the reference angle data and the actual measured angle data, the computer will be able to calculate the amount of deviation that exists from the reference translunar trajectory.

The prerequisites of this function is the establishment and maintenance of a primary inertial reference, the performance of an attitude hold mode, and the performance of controlled rotation to specified attitudes. All major G and N components with their normal controls and displays would then be required to perform this function.

<u>Translunar Trajectory Miss - Distance</u> - The procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the projected miss-distance, obtain a GOSS determination of the miss-distance, compare the on-board data with the GOSS determined data, and then determine the final estimated miss-distance of the present translunar trajectory at the key lunar aiming point.

The on-board procedure assumes that the spacecraft computer will have been provided with four body equations of motion that define a reference translunar trajectory. The spacecraft computer will then accept an actual translunar trajectory data input and on the basis of the difference between the actual trajectory and the reference trajectory will be able to calculate the miss-distance of the present trajectory at the key lunar aiming point.

Only the AGC with its normal controls and displays will be required to perform this system function.

Translunar Mid-Course Correction Parameters - The procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the parameters of the desired mid-course correction, obtain a GOSS determination of the mid-course correction, compare the on-board data with the GOSS determined data and then determine the final corrected parameters of the mid-course correction.

The on-board procedure will be to insert the present trajectory data and the projected miss-distance data into the spacecraft computer and then initiate a computer calculation that will determine the injection time and place and the desired mid-course thrust correction that is required to correct the miss-distance and hit the specified key lunar aiming point.

Only the AGC with its normal controls and displays will be required to perform this system function.

G and N Large Δ V Mode - This integrated function will consist of the large Δ V changes that are directed by G and N signals and are controlled by the SCS control and display signals and which then result in the appropriate rotational impulse and thrust impulse as supplied by the Service Module Reaction Control System and the Service Module Propulsion System.

The G and N prerequisites of this large Δ V are to determine the appropriate injection or mid-course correction parameters, establish and maintain a primary inertial reference, generate signals for a G and N Attitude Hold Mode, and generate signals for a controlled rotation to a specified thrust attitude.

The preliminary space crew tasks of this large Δ V are to insert specific commands into the AGC, set and start the event time clocks, set the required Δ V and the engine tail-off correction into the SCS Δ V display panel, determine the present mass of the spacecraft and the present CG location, and set the appropriate gimbal angles into the SCS gimbal control and position indicator panel. The three attitude dead band control selectors would then be positioned to minimum or $\pm 0.5°$ dead band and finally the SCS control selector would be set on the G and N Δ V mode.

The space crew would then watch the event time clock as it goes to zero and at this time would manually initiate the ullage acceleration by commanding + X translation on the SCS translational controller.The accelerometers on the IMU stable element will sense the ullage acceleration and when a specified velocity change is detected the AGC will send an engine fire signal to the SCS and the service module propulsion engine. This signal will initiate engine firing and will be displayed by illuminating a light behind the engine fire push button on the SCS Δ V display panel. The space crew would then stop commanding + X translation and would monitor the Δ V remaining indication and the 150 feet/ second meter on the SCS Δ V display panel. When the Δ V remaining value passes through the selected tail-off correction the AGC will sense this ΔV and will then send an engine cut-off signal to the SCS and the service module propulsion engine. This signal will initiate engine cut-off, will turn-off the light behind the engine fire push button and will illuminate a light behind the engine cut-off push button on the SCS ΔV display panel. The G and N large ΔV would then be complete and the space crew may then select a G and N or a SCS Attitude Hold Mode on the SCS control panel.

During a G and N large ΔV mode and before engine firing the roll, pitch and yaw attitude control signals that are generated by the G and N system will only go to the Service Module Reaction Control System. After engine firing and until engine cut-off the pitch and yaw attitude control signals go only to the pitch and yaw gimbals of the service propulsion engine and the roll attitude control signals go only to the Service Module Reaction Control System. During a G and N large ΔV Mode and after cut-off the roll, pitch and yaw attitude control signals that are generated by the G and N system will only go to the Service Module Reaction Control System.

The prerequisites of this G and N ΔV function are as described above. The G and N components that would then normally be required by this function are the IMU, PSA, AGC, CDU, and their respective controls and displays.

<u>ON-OFF Thrust Display Signals for G and N Small ΔV</u> - A small ΔV is defined as a specified + X thrust that is obtained by the Service Module Reaction Control System. This particular integrated function will

Page 130 of the report has been deleted.

consist of the maintenance of a G and N Attitude Hold Mode while the space crew operates the manual translational controls of the SCS and directs the Service Module Reaction Control System to perform a specified + X small Δ V.

The G and N part of this integrated system function is to maintain a G and N attitude hold mode and generate thrust ON-OFF signals that are displayed to the space crew.

The G and N prerequisites of this small Δ V are to determine the mid-course correction parameters, establish and maintain a primary inertial reference, generate signals for a G and N attitude hold mode, and generate signals for a controlled rotation to a specified thrust attitude.

The preliminary space crew tasks of this small Δ V are to insert specific commands into the AGC, set and start the event time clocks, set the required Δ V into the SCS Δ V display panel and set the minimum or $\pm$ 0.5° dead band control setting on all three dead band control selectors. The G and N Attitude Hold Mode is then selected and maintained during this function.

The space crew would then watch the event time clock as it goes to zero and at this time the G and N AGC will generate a space crew thrust-ON signal that will be displayed by illuminating a light behind the engine-fire push button on the SCS Δ V display panel. The space crew would use this light as a signal to operate the SCS manual + X translational controller to direct the reaction control system to provide + X thrust.

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The G and N prerequisites of this small ΔV are to determine the mid-course correction parameters, establish and maintain a primary inertial reference, generate signals for a G and N attitude hold mode, and generate signals for a controlled rotation to a specified thrust attitude.

The preliminary space crew tasks of this small ΔV are to insert specific commands into the AGC, set and start the event time clocks, set the required ΔV into the SCS ΔV display panel and set the minimum or $\pm 0.5^{\circ}$ dead band control setting on all three dead band control selectors. The G and N Attitude Hold Mode is then selected and maintained during this function.

The space crew would then watch the event time clock as it goes to zero and at this time the G and N AGC will generate a space crew thrust-ON signal that will be displayed by illuminating a light behind the engine-fire push button on the SCS ΔV display panel. The space crew would use this light as a signal to operate the SCS manual +X translational controller to direct the reaction control system to provide +X thrust. The accelerometers on the IMU stable element will sense the specified ΔV and when the ΔV remaining indicates zero the AGC will generate a space crew thrust-OFF signal that will turn-off the light behind the engine-fire push button and will illuminate a light behind the engine cut-off push button on the SCS ΔV display panel. The space crew would then stop commanding +X translation and the G and N small ΔV function would be completed.

[redacted]

The prerequisites of this G and N small ΔV function are as described above. The G and N components that would then normally be required by this function are the IMU, PSA, AGC, and their respective controls & displays.

Lunar Orbit Injection Parameters - The procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the lunar orbit injection parameters, obtain a GOSS determination of the injection parameters, compare the on-board data with the GOSS determined data and then determine the final corrected values of the lunar orbit injection program.

The on-board AGC computations assume that the AGC will have been provided with four body equations of motion that define a reference trans-lunar trajectory, and a reference lunar orbit injection program. The AGC will then accept actual translunar trajectory data and on the basis of the difference between the actual and the reference trajectory data the AGC will be able to calculate the injection time and place and the injection parameters that will be required to obtain a near reference lunar orbit.

Only the AGC with its normal controls and displays will be required to perform this system function.

Lunar Orbit and Ephemerides - The procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the lunar orbit trajectory, obtain a GOSS determination of the orbit trajectory, compare the on-board data with the GOSS determined data and then determine the final corrected values of the lunar orbit trajectory.

The on-board G and N procedure will be to perform a G and N attitude hold procedure, generate signals to maneuver the spacecraft to specified attitudes, perform a series of SCT and SXT navigational sightings on known landmarks and stars, insert the navigational sighting data into the AGC, initiate computer programs that will first smooth and average out the various navigational sightings and will then compute the present lunar orbit trajectory and predict the ephemerides of future lunar orbits.

The prerequisites of this G and N function is the establishment and maintenance of a primary inertial reference, the performance of a G and N attitude hold mode, and the performance of controlled rotation to specified attitude. All major G and N components with their normal controls and displays would then be required to perform this function.

LEM G and N Support - The details and requirements of this support function are not known at this time.

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Transearth Injection Parameters - The procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the transearth injection parameters, obtain a GOSS determination of the injection parameters, compare the on-board data with the GOSS determined data and then determine the final corrected values of the transearth injection program.

The on-board AGC computations assume that the AGC will have been provided with four body equations of motion that define a reference lunar orbit and a reference transearth injection program. The AGC will then accept actual lunar orbit data and on the basis of the difference between the actual and the reference lunar orbit data the AGC will be able to calculate the injection time and place and the injection parameters that will be required to hit the specified key earth aiming point.

Only the AGC with its normal controls and displays will be required to perform this system function.

Present Transearth Trajectory - The procedural description, the prerequisites and the requirements for G and N components of this system function are the same as the function that is listed under Present Translunar Trajectory.

Transearth Trajectory Miss-Distance - The procedural description and the G and N components that are required for this system function are listed under Translunar Trajectory Miss-Distance.

Transearth Mid-Course Correction Parameters - The procedural description and the G and N components that are required for this system function are listed under Translunar Mid-Course Correction Parameters.

Earth Entry Parameters - The general procedural description of this system function is as follows:

The general procedure will be to perform an on-board determination of the desired entry parameters, obtain a GOSS determination of the entry parameters, compare the on-board data with the GOSS determined data, and then determine the final corrected values of the time and place of entry and the atmosphere entry program.

The on-board AGC computations assume that the AGC will have been provided with four body equations of motion that define a reference transearth trajectory, a reference key aiming point, a reference atmosphere entry program and a specified landing area. The AGC will then accept actual transearth trajectory data and on the basis of the difference between the actual and the reference trajectory data the AGC will be able to calculate the entry attitude, time, and place and the entry parameters that will be required to perform a specified atmosphere entry program and touchdown in the desired landing area.

[redacted]

Only the AGC with its normal controls and displays will be required to perform this system function.

G and N Entry Mode - This integrated function consists of the attitude hold and attitude roll maneuvers that will be programmed and directed by G and N signals, controlled by SCS control and display signals, and rotated and stabilized by the impulse of the Command Module Reaction Control System. The G and N entry mode will normally be used during command module entry from approximately 400,000 feet to about 50,000 feet and should not be selected until after service module separation and the final entry attitude orientation has been made.

The G and N prerequisite of this entry function are to determine the earth entry parameters, establish and maintain a primary inertial reference, generate signals for a G and N attitude hold mode, and generate signals for a controlled rotation to a specified entry attitude.

The preliminary space crew tasks of this entry function are to insert specific entry commands into the AGC, set and start the event time clocks, select the G and N Entry Mode and then monitor the SCS entry corridor display and FDAI since the controlled maneuvers in the normal G and N entry mode are fully automatic.

The G and N entry mode will maintain a specified entry attitude until atmospheric drag occurs and will thenswitch to the programmed entry maneuvers when the accelerometers on the IMU stable element senses a drag force of .05g. The G and N system will generate the attitude displacement and error signals that will be displayed on the SCS FDAI. The G and N system will also control and program the entry maneuvers and will generate the g load versus time-data and the data on the offset CG pitch axis roll angle which will be displayed on the SCS entry corridor display panel.

The prerequisites of this G and N entry function are described above. The G and N components that would then normally be required by this function are the IMU, PSA, AGC, CDU, and their respective controls and displays.

STABILIZATION AND CONTROL SYSTEM

The Apollo Stabilization and Control System is a manual and a semi-automatic spacecraft stabilization and control system that is directed and operated by the space crew to provide the SCS display and control signals that are required by the space crew, the Guidance and Navigation Control System, and the Command Module Reaction Control System.

During the initial phases of a lunar landing mission, when the S-IVB is attached to the spacecraft, the Apollo Stabilization and Control System will only display attitude and flight path monitoring data to the space crew. After translunar injection and after the S-IVB stage has been separated from the spacecraft, the Apollo Stabilization and Control System will then perform all of the SCS functions and generate all of the SCS control and display signals that are required by the spacecraft to complete the lunar landing mission.

The Stabilization and Control System will include the following major components:

Body Mounted Attitude Gyros (BMAG)

Rate Gyro Package (RGP)

Euler Angle Generator (EAG)

X Axis Accelerometer

SCS Electronics

SCS Control and Display

The SCS control and display components and functions will include: SCS control panel, flight director attitude indicator (FDAI), gimbal position indicator, delta velocity ΔV indication, entry corridor display, manual attitude control, emergency manual attitude control, manual translational control, emergency translational control, manual attitude error control, manual proportional rate attitude control, manual BMAG drift trim, manual beadband adjust, manual ΔV command, and clock timer indicatior.

The space crew and the Stabilization and Control System performs the following pertinent functions on a normal lunar landing mission.

<u>Secondary Inertial Reference</u> - A secondary attitude inertial reference will be established and maintained by the SCS. The mode of reference framework may be selected with respect to a number of coordinate axes. The SCS inertial reference may be shut down when it is not needed and new inertial references may be established, as required, during the mission.

The SCS components that are normally required by the maintenance of this secondary attitude inertial reference are the BMAG, the RGP, the EAG, the SCS electronics and their normal controls and displays.

The establishment of a secondary inertial reference, on the launch pad, prior to take-off, will require the SCS components listed above plus some special aerospace support equipment and a ground alignment procedure.

The establishment of a secondary inertial reference, during the mission will require the SCS components listed above plus G and N components including the IMU, the SCT, the SXT and their normal controls and displays.

<u>Attitude Rate-of-Change</u> - Spacecraft attitude rate-of-change signals will be generated by the rate gyro package and these signals will be displayed to the crew by the FDAI and sent to the SCS electronics package as rate commands and rate error or rate dampen commands. This SCS function is the only way that attitude rate-of-change data is displayed to the space crew and it will normally be used full time or part time on each mission phase.

The SCS components that are normally required by this function are, the RGP, the FDAI, the SCS electronics and their normal controls and displays.

<u>SCS Monitor Mode</u> - In the SCS monitor mode the SCS will generate attitude rate-of-change data and display this data on the FDAI when the spacecraft rotates around an x, y, or z axis. No control signals will be transmitted by the SCS to the attitude or thrust control systems in this monitor mode. The G and N system will normally provide a primary inertial reference and will provide attitude displacement and error signals that will also be displayed by the SCS FDAI during this monitor mode.

The prerequisite of this function is the establishment and maintenance of a primary and a secondary inertial reference. The SCS components that would then normally be required by this function are the RGP, the FDAI, the SCS electronics package and their normal controls and displays.

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SCS Attitude Hold Mode - Attitude hold will be the normal spacecraft function of this mode. An adjustable deadband for the different functional requirements of attitude hold is available and may be selected by channel on the SCS control panel. Attitude displacement (control and display) signals will be generated by the EAG and BMAG to drive the ball attitude indicator of the FDAI and control the spacecraft attitude by directing the appropriate Reaction Control System. Attitude error display signals will be generated by the BMAG and displayed by the FDAI and will indicate the attitude displacement within the selected deadband setting.

A prerequisite of this function is the establishment and maintenance of a secondary inertial reference. The SCS components that would then normally be required by this function are the BMAG, the RGP, the EAG, the FDAI, the SCS electronics package and their normal controls and displays.

Provide Signals and Displays for a G and N Attitude Hold Mode - G and N attitude hold will be the normal spacecraft function of this mode and the operational procedure of this mode will be described in of the G and N pertinent function section under G and N Attitude Hold Mode.

Attitude rate signals, for this mode will be generated by the RGP and displayed to the space crew on the FDAI. Attitude displacement and error signals, for this mode, will be generated by the G and N System and displayed to the space crew on the SCS FDAI.

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The SCS prerequisite of this function is the initiation and operation of the RGP. The SCS components that would then normally be required are: the FDAI, the RGP, the SCS electronics, plus the normal controls and displays of the SCS components.

SCS Local Vertical Mode - The SCS local vertical mode is a special case of orbital attitude hold using the SCS system. No attitude maneuver capability exists in this mode. This function will normally be used during the orbital phases to hold the spacecraft pitch axis at some fixed angle with respect to the local vertical of a near body.

A prerequisite of this function is the establishment and maintenance of a secondary inertial reference. The SCS components that would then normally be required by this function are: the orbital rate package, the BMAG, the RGP, the EAG, the FDAI, the SCS electronics and their normal controls and displays.

Controlled Rotation to Specified Attitudes - This function will consist of the manually controlled attitude maneuvers that are directed by the SCS system during a SCS attitude hold mode or a G and N attitude hold mode. This manually controlled function will include the capability of commanding specific attitudes as well as specified rates of change of attitudes. Two right hand rotational controllers are included in the SCS equipment to perform this function.

A prerequisite of this function is the establishment and maintenance of a SCS or G and N Attitude Hold Mode. The normal procedure will then be for the space crew to monitor the attitude ball of the FDAI or the celestial or near body references and direct the spacecraft to the specified attitude by the proper movement of one of the right hand rotational controllers.

Each right hand rotational controller will have an emergency switch which may be engaged at any time to create an emergency mode which results in the direct control of the appropriate attitude control jets.

A prerequisite of this normal function is the establishment and maintenance of a SCS or G and N Attitude Hold Mode. The SCS components that would then be required are: the BMAG, the RGP, the EAG, the FDAI, the SCS electronics, one of the right hand rotational controllers, plus the normal controls and displays of the SCS components.

Free Drift or Rotation Around an Axis - The free drift function is a special capability of a SCS or G and N Attitude Hold Mode. If a free drift or rotation is desired in any channel, during an attitude hold, the channel disable control on the SCS control panel can be activated. In this condition the output signals to the appropriate reaction control system are inhibited.

A prerequisite of this function is the establishment and maintenance of a SCS or G and N Attitude Hold Mode. The SCS components that would then normally be required are the BMAG, the RGP, the EAG, the FDAI, the SCS electronics, the SCS Control Panel and the normal controls and displays of the SCS components.

SCS Large ΔV Mode - This integrated function will consist of the large ΔV changes that are initiated and directed by the space crew and controlled by the SCS control and display signals which result in the appropriate rotational impulse and thrust impulse of the Service Module Reaction Control System and the Service Module Propulsion System.

The space crew and the G and N prerequisite of this SCS large ΔV are to determine the appropriate injection or mid-course correction parameters. The space crew and the SCS prerequisites of this large SCS ΔV are to establish and maintain a secondary inertial reference, provide a SCS Attitude Hold Mode, and generate signals for a controlled rotation to a specified thrust attitude.

The preliminary space crew tasks of this SCS large ΔV are to set and start the event time clocks, set the required ΔV and the engine tail-off correction into the SCS ΔV display panel, determine the present mass of the spacecraft and the present CG location and set the appropriate gimbal angles into the SCS gimbal angle control and position indicator panel. The three attitude deadband control selectors would then be positioned to a minimum or $+0.5^{\circ}$ deadband and finally the SCS control selector would be set on the SCS ΔV mode.

The space crew would then watch the event time clock as it goes to zero and at this time would manually initiate the ullage acceleration by commanding +X translation on the SCS translational controller. The space crew would then watch the elapse time or the velocity change displayed and will manually give the engine fire signal by operating the engine fire control on the SCS ΔV display panel.

This signal will initiate engine firing and will be displayed by illuminating a light behind the engine fire push button on the SCS ΔV display panel. The space crew would then stop commanding +X translation and would monitor the remaining indication and the 150 feet/second meter on the SCS ΔV display panel. When the ΔV remaining value passes through the selected tail-off correction a semi-automatic engine cut-off signal will be initiated by the SCS X-axis accelerometer package. This signal will initiate engine cut-off, will turn-off the light behind the engine fire push button and will illuminate a light behind the engine cut-off push button on the SCS ΔV display panel. The SCS large ΔV is now complete and the space crew may now select the SCS Attitude Hold Mode on the SCS control panel.

During a SCS large ΔV mode and before engine firing the roll, pitch and yaw attitude control signals that are generated by the SCS will only go to the Service Module Reaction Control System. After engine firing and until engine cut-off the pitch and yaw attitude control signals go only to the pitch and yaw gimbals of the service propulsion engine and the roll attitude control signals go only to the Service Module Reaction Control System. During a SCS Large ΔV Mode and after engine cut-off the roll, pitch and yaw attitude control signals that are generated by the SCS will only go the the Service Module Reaction Control System.

The prerequisite of this SCS large ΔV function are as described above. The SCS components that would then normally be required by this function are: the BMAG, the RGP, the EAG, the FDAI, the SCS electronics, the SCS Control panel, the SCS ΔV Panel, the Gimbal Position Control Panel, the manual rotational controller, and the manual translational controller plus the normal controls and displays of the SCS components.

G and N Large ΔV Mode - This integrated function will consist of the large ΔV changes that are directed by G and N signals and are controlled by SCS control and display signals and which then result in the appropriate rotational impulse and thrust impulse as supplied by the Service Module Reaction Control System and the Service Module Propulsion System.

The normal operational procedure of this integrated function is described in the G and N pertinent functions section under G & N Large ΔV Mode.

The SCS components that would then normally be required by this function are: the BMAG, the RGP, the EAG, the FDAI, the SCS electronics, the SCS Control Panel, the SCS ΔV Panel, the Gimbal Position Control Panel, the manual translational controller plus the normal controls and displays of the SCS components.

SCS Small ΔV and Translation Thrust - A small ΔV is defined as the specified +X thrust that is obtained by the Service Module Reaction Control System. This particular SCS function will consist of the maintenance of an SCS Attitude Hold Mode while the space crew operates the manual translational controls and directs the Service Module Reaction Control System to perform a specified +X small ΔV.

The space crew and the G and N prerequisites of this small ΔV are to determine the appropriate mid-course correction or orbital correction parameters. The space crew and the SCS prerequisites of this small ΔV are to establish and maintain a secondary inertial reference, provide a SCS Attitude Hold Mode, and generate signals for a controlled rotation to a specified thrust attitude.

The preliminary space crew tasks of this small ΔV are to set and start the event time clocks, set the required ΔV into the SCS ΔV display panel, and set the minimum or $\pm 0.5^{\circ}$ deadband control setting on all three deadband control selectors. The SCS Attitude Hold Mode is then selected and maintained during this function.

The space crew would then watch the event time clock as it goes to zero and at this time would operate the manual +X translational controller to direct the reaction control system to provide a +X thrust. The ΔV control panel will display the ΔV remaining and when the ΔV remaining indicates zero the space crew will then stop commanding +X translation and the SCS small ΔV function would be complete.

The SCS manual translational controller may also be used to generate manual control signals which will direct the Service Module Control System to produce translational forces along the y and z axes. This particular capability uses visual references, a minimum or ± 0.5 degree position of the deadband adjustment on each channel and a G and N or SCS Attitude Hold Mode. y and z translations are commanded only by the SCS manual translational controller and these SCS small ΔV's are not sensed and indicated on the SCS ΔV display panel.

<u>ON-OFF Thrust Signals for G and N Small ΔV Display</u> - The normal operational procedure of this integrated function is as described in of the G and N pertinent functions under ON-OFF Thrust Display Signals for G & N Small ΔV.

The SCS part of this integrated function will be to first display a space crew thrust - ON signal that will be displayed by illuminating a light behind the engine fire push button on the SCS ΔV Display Panel. The second part of this SCS function will be to display a space crew thrust - OFF signal that will be displayed by turning off the light behind the engine-fire push button and will illuminate a light behind the engine but-off push button of the SCS ΔV Display Panel. The space crew would then stop commanding +X translation and the G and N Small ΔV function would be complete.

The SCS components that will then normally be required by this function are the SCS Control Panel, the SCS event time clocks, the SCS ΔV Control Panel, and the SCS manual translational controller plus the normal controls and displays of the SCS components.

<u>X-Axis Velocity Data</u> - This pertinent function is a special capability of the SCS ΔV X-axis accelerometer package and the SCS ΔV Display Panel. The X-axis accelerometer package will include a body mounted accelerometer and an integrator. The accelerometer will be able to measure the spacecraft accelerations that result from +X thrust and the aerodynamic re-entry aero-dynamic re-entry (wind-axis) drag forces. The integrator will then be able to generate velocity signals which are subtracted from the ΔV required data to displayed on the ΔV remaining portion of the ΔV Display Panel. A special

capability of this pertinent function is the semi-automatic generation of the engine cut-off signal during a SCS Large ΔV Mode as previously shown. The ΔV remaining data that is displayed on the ΔV display panel will normally be used by the crew to monitor both of the large ΔV operational procedures and as space crew thrust On and thrust OFF manual control indicator for both of the small ΔV operational procedures.

The prerequisite of this function is the initial input of ΔV required and ΔV remaining and the setting of the accelerometer integrator to zero. The SCS components that would then normally be required are the body mounted accelerometer package and the ΔV Control Panel.

Time Data - The time-to-go- and time-from-event function is a special capability of the event time clocks.

The prerequisites of this function are the initial input and calibrated start of the event time clocks. Only the event time clocks and space crew monitoring will then be required by this function.

SCS Entry Mode - This integrated function consists of the SCS display of attitude, g-load and entry corridor data that is generated by the G and N system and the SCS; and the space crew operation of the SCS manual roll controller which will then result in the appropriate rotational and stabilizing impulse as supplied by the Command Module Reaction Control System.

The rotational control of this SCS Entry Mode is a completely manual operation; however, the pitch and yaw channels are switched to an automatic SCS rate damping mode when the accelerometers on the IMU stable element sense aerodynamic drag forces. The SCS Entry Mode may be used during a Command Module entry from approximately 400,000 feet to about 50,000 feet and this mode should not be selected until after Service Module separation and after the final entry attitude orientation has been made.

The space crew and the G and N prerequisites of this function are to determine the earth entry parameters, and establish and maintain a primary inertial reference.

The space crew and the SCS prerequisites of this function are to establish and maintain a secondary inertial reference, provide a reference input and initiate the operation of the SCS Entry Corridor Display Panel, provide a SCS Attitude Hold Mode, generate signals for a controlled rotation to a specific entry attitude, and then to select the SCS Entry Mode.

The SCS Entry Mode will maintain the specified entry attitude until atmospheric drag occurs and will switch pitch and yaw channels to an automatic SCS rate damping mode when the accelerometers on the IMU stable element sense a drag force of .05g. The G and N system will generate g load versus time data and data on the offset CG pitch axis roll angle which will be displayed on the SCS Entry Corridor Display Panel. The SCS will generate the attitude displacement, attitude error, and attitude rate signals which will be displayed on the FDAI. The space crew will monitor the FDAI and the Entry Corridor Display Panel and will operate the SCS manual rotational controller and control the offset CG pitch axis roll angle to a safe and appropriate roll attitude by manually controlling the Command Module Reaction Control System.

The prerequisite of this SCS entry function are as described above. The SCS components that would then normally be required by this function are: the BMAG, the RGP, the EAG, the SCS electronics, the FDAI, the SCS Entry Corridor Display Panel, the SCS manual rotational controller, plus the normal controls and displays of the SCS components.

G and N Entry Mode - This integrated function consists of the attitude hold and attitude roll maneuvers that will be directed by G and N generated signals, controlled and displayed by SCS control and display signals and which will then result in the appropriate rotational and stabilizing impulse as supplied by the Command Module Reaction Control System.

The operational procedure of this function is as described in the G and N pertinent function section under G & N Entry Mode.

The SCS prerequisite of this G and N entry function are to provide a reference output and initiate the operation of the SCS Entry Corridor Display Panel. The SCS components that would then normally be required by this function are: the BMAG, the RGP, the SCS electronics, the FDAI, the SCS Entry Corridor Display Panel plus the normal controls and displays of the SCS components.

SERVICE MODULE REACTION CONTROL SYSTEM

The Service Module Reaction Control System provides impulse for both attitude control and small translational velocity changes. The system operates in response to control signals generated by the SCS or a secondary electrical circuit connected to manual override controls. After the spacecraft is separated from the S-IVB booster stage, the S/M Reaction Control System is used to accomplish docking, small trajectory velocity changes, and orientation maneuvers.

The Service Module Reaction Control System consists of four similar systems, each independent of the remaining three. Each of these identical systems consists of the following subsystems:

- Helium Storage and Distribution
  - Helium Pressure Vessel
  - Manual Helium Valve
  - Manual Helium Solenoid Valve
  - Helium Pressure Regulators
  - Helium Pressure Check Valves
  - Helium Pressure Relief Valves
  - Manual Vent Valves
- Oxidizer Storage & Distribution
  - Oxidizer Tank
  - Manual Oxidizer Fill Valve
  - Solenoid Operated Valve
- Fuel Storage & Distribution
  - Fuel Tank
  - Manual Fuel Fill Valve
  - Solenoid Operated Valve
- Four Engine Cluster
  - Propellant Valves
  - Thrust Chamber Assemblies

The four similar systems provide a total of 16 reaction control engines - 8 roll, 4 pitch, and 4 yaw. The system is a pressure-fed, bi-propellant liquid type with non-throttable, radiation/ablative cooled engines that have solenoid propellant valves which are pulse modulated for impulse control.

The space crew and the Service Module Reaction Control System perform the following pertinent functions during a normal lunar landing mission.

ATTITUDE & TRANSLATIONAL IMPULSES

For each maneuver electrical signals are sent to selected RCS engines to provide the required vectored impulse. Both propellant valves of each selected RCS engine open simultaneously upon receipt of the electrical signal and remain open until cessation of the signal. Opening of the propellant valve allows pressure regulated helium from the high pressure helium storage vessel to flow into the ullage side of positive expulsion diaphragm in the propellant tanks and displace propellants. The displaced propellants are distributed to the RCS engine combustion chamber through the propellant valves. The hypergolic propellants spontaneously ignite upon mixing and the combustion products are directed through a nozzle and out of the thrust chamber thus producing a thrust reaction. Components primarily involved in this pertinent function have been listed.

COMMAND MODULE REACTION CONTROL SYSTEM

The Command Module Reaction Control System provides impulse for attitude control. The system operates in response to control signals generated by the SCS or a secondary electrical circuit connected to manual override controls. Subsequent to jettisoning the Service Module, the system is used to orient the spacecraft for entry and to accomplish roll maneuvers necessary for controlling entry parameters.

The Command Module Reaction Control System consists of two similar but independent systems, one of which may provide adequate control if necessary. Each of these systems consists of the following subsystems:

Helium Storage & Distribution

Helium Pressure Vessel
Manual Helium Valve
Helium Squib Valve
Manual Helium Solenoid Valve
Helium Pressure Check Valves
Helium Pressure Relief Valves
Manual Vent Valves

Oxidizer Storage & Distribution

Oxidizer Tank
Manual Oxidizer Fill Valve
Oxidizer Burst Diaphragm
Manual Oxidizer Solenoid Valve

Fuel Storage & Distribution

Fuel Tank
Manual Fuel Fill Valve
Fuel Burst Diaphragm
Manual Fuel Solenoid Valve
Engine
Propellant Valves
Thrust Chamber Assemblies

The two similar systems provide a total of 12 reaction control engines - 4 roll, 4 pitch, and 4 yaw. The system is a pressure-fed, bi-propellant liquid type with non-throttable, radiation/ablative cooled engines that have solenoid propellant valves which are pulse modulated for impulse control.

The space crew and the Command Module Reaction Control System perform the following pertinent functions during a normal lunar landing mission:

Initial Pressurization Sequence -

After the S/M is jettisoned the RCS propellant tanks are pressurized prior to operation of the RCS. The pressurization is initiated either by an automatic signal from the SCS or through manual controls. The electric command signal opens a pyrotechnic valve which releases high pressure helium from the helium storage vessel to the downstream pressure regulators and propellant tanks. The displaced propellants in turn rupture the downstream burst diaphragms and will then flow to selected thrust chambers when their NC propellant valves open. Components primarily involved in this function are the helium storage vessel, helium squib valve, helium pressure regulator, helium pressure check valves, oxidizer tank, fuel tank, oxidizer burst diaphragm and fuel burst diaphragm.

Attitude Impulse

For required vectored impulse selected pairs of RCS thrust chamber propellant valves are open by an electrical signal and remain open until cessation of the signal. Pressure-fed propellants flow through the propellant valves into the thrust chamber combustion area. Upon mixing, the hypergolic propellants spontaneously ignite and the combustion products flow through a nozzle and out of the thrust chamber thus producing a thrust reaction. Components primarily involved in this pertinent function are propellant valves and thrust chamber assemblies.

[redacted]

SERVICE PROPULSION SYSTEM

The Service Propulsion System provides impulse for large spacecraft vector velocity changes. The system operates in response to electrical signals generated by the SCS or by manual override controls provided for the crew. After LEM transposition, the Service Propulsion System is operated for midcourse corrections, injection into lunar orbit, lunar orbit changes, and transearth injection.

The Service Propulsion System consists of the following subsystems:

Helium Storage and Distribution

Helium Pressure Vessel
Manual Helium Fill & Drain Disconnect
Helium Pressure Coupling
Solenoid Operated Valves
Helium Pressure Regulators
Helium Pressure Check Valves
Helium Pressure Relief Valves

Propellant Storage and Distribution

Manual Oxidizer Vent Disconnect
Oxidizer Tanks
Manual Oxidizer & Drain Disconnect
Propellant Utilization Valve
Manual Fuel Vent
Fuel Tanks
Manual Fuel Fill & Drain Disconnect

Rocket Engine

Thrust Chamber Assembly
Propellant Valve Assembly
Disconnect Assembly

Gimbal

Servo Actuator Assembly
Thrust Gimbal Ring Assembly

 [redacted]

[redacted]

The space crew and the Service Propulsion System perform the following pertinent functions during a normal lunar landing mission:

Thrust Impulse

The main propellant valve opens in response to an electrical signal, and remains open until the cessation of the signal. Opening of the propellant valve allows pressure regulated helium from the high pressure helium pressure vessel to flow into the propellant tanks and displace propellants. The displace propellants flow through the main propellant valve into the rocket engine combustion chamber through an injector. The hypergolic propellants spontaneously ignite upon mixing and the combustion products are directed through a de Laval nozzle and out of thrust chamber thus producing a thrust reaction.

The components primarily involved in this function are: the helium pressure vessel, 2 primary helium pressure regulators and 2 secondary helium pressure regulators, 8 helium pressure check valves, oxidizer tanks,fuel tanks,propellant valve assembly, and the thrust chamber assembly.

[redacted]

Propellant Utilization & Flow Ratio Adjustment

Propellant Utilization valve varies the oxidizer flow within limits to adjust the fuel/oxodizer ratio flow to the rocket engine. The purpose of this function is to deplete the fuel and oxidizer at the same time. The propellant valve may be controlled either manually or automatically through use of fuel quantity sensors in the propellant tanks. The components primarily involved in this function are the propellant utilization valve and the propellant quantity sensors.

Gimbal Operation and Angle Presetting

Propellant consumption and S/C configuration changes move the center of gravity. To compensate for CG shifts, the gimbal actuator motors are turned on and the thrust chamber gimbal angle is preset manually. Also, prior to operation of the propulsion system, the gimbal motors must have been brought up to speed. When the propulsion system is in operation, the gimbal subsystem is controlled through the stabilization and control system by the guidance and navigation subsystem to maintain or program a directional thrust. The components primarily involved in this function are 2 servo actuator assemblies and the thrust gimbal ring assembly.

ENVIRONMENTAL CONTROL SYSTEM

The Environmental Control System provides the crew with a controlled environment necessary for crew comfort, safety and optimum operation of equipment during the Apollo mission. In addition to providing both a pressure suit and a "shirtsleeve" atmosphere, the Environmental Control System provides a suitable temperature environment for the spacecraft equipment. It also provides a replenishing outlet for self-contained extra-vehicular pressure support systems (back packs). The Environmental Control System also provides water for crew consumption and heat transfer operations.

The Environmental Control System consists of the following subsystem and related major components:

Pressure Suit

- Debris trap
- Suit compressor
- CO_2 & odor absorbers
- Regenerative heat exchanger
- Suit water evaporator
- Glycol-to-suit air heat exchanger
- Water separator

Water-glycol

- Glycol pump
- Glycol evaporator
- Glycol reservoir

Command Module Pressure and Temperature Control

- Cabin heat exchanger
- Blower
- Cabin pressure regulator and negative relief valve
- Inflow and outflow manual control valves

Oxygen Supply

Normal oxygen supply
Entry oxygen supply
Back-pack oxygen supply

Water Supply

Potable water tank
Water chiller
Waste water tank

Waste Management

Urine and fecal receptable
Vacuum cleaner
Urine separator
Germicide tank
Blower

The space crew and the Environmental Control System perform the following pertinent functions during a normal lunar landing mission:

Pressure Suit Environment - The Environmental Control System provides a pressure suit environment for all mission phases. This function, occurring while the crewmen are in their pressure suits, consists of the interaction of the following steady-state sub-functions:

Circulation of oxygen - The pressure suit circuit provides the crew with oxygen during pressure suit or shirtsleeve operation. The pressure required for the oxygen flow is supplied from the pressurized oxygen storage tanks located in the service module, during all mission phases except entry and parachute descent. After the service module has been jettisoned, the oxygen is supplied by the entry oxygen supply. Two pressure suit compressors are also used to circulate the gas flow through the pressure suit circuit. The normal oxygen supply has the additional capability of recharging the individual backpacks, which are used for extra-vehicular activity.

This subfunction requires the following equipment:

Regular oxygen storage tanks

Entry oxygen storage tanks

Suit compressors

• <u>Oxygen Flow Temperature Control</u>

In order to maintain the temperature of the oxygen flow within operational limits, the oxygen is circulated through a regenerative heat exchanger and an integrated heat exchanger package. The oxygen flow enters the hot side of the regenerative heat exchanger, where it is cooled slightly by the dry, dehumidified pressure suit gas, passing through the cold side of the heat exchanger. The gas next flows through the integrated heat exchanger package which consists of a suit evaporator and a glycol-to-suit air heat exchanger. The integrated heat exchanger package removes heat from the circulating gas and condenses excess moisture so that it can subsequently be removed from the gas flow.

[redacted]

Operating in conjunction with the pressure suit circuit, which contains the heat exchanger package is the water-glycol subsystem which acts as a heat sink for the pressure suit circuit in that it supplies the heat transport medium to the heat exchangers. Cold water-glycol is directed through the glycol-to-suit air heat exchanger, where the pressure suit circuit heat load is absorbed. Ordinarily, the cold water-glycol which always flows through the integrated heat exchanger package can cool the circulating gas to 50°F. If however, the temperature at the package discharge end exceeds the desired temperature, as sensed by a temperature sensor, sufficient water is supplied to the suit evaporator to maintain 50°F in the package discharge gas.

During the ascent phase the water-glycol bypasses the service module and the space radiators are kept dry to protect them against the high temperatures encountered by aero-dynamic heating in the lower atmosphere. When the vehicle reaches approximately 200,000 feet, the water-glycol evaporator is activated to cool the circulating water glycol. Upon achieving orbit water-glycol is supplied to the space radiators.

[redacted]

This sub-function requires the following equipment:

Regenerative heat exchanger
Suit evaporator
Glycol-to-suit air heat exchanger
Space radiators
Glycol evaporator
Waste water supply

Purification of C/M Atmosphere & Oxygen Flow

Purification of the pressure suit circuit O_2 flow refers to the elimination and control of foreign matter, CO_2 and odors, and water vapor. Contamination of the gas flow by particles or foreign matter is eliminated by circulating the gas flow through a debris trap which filters out contaminants before the gas enters the suit compressors. Carbon dioxide and odors are removed from the gas flow by two CO_2 and odor absorbers which contain LIOH to absorb the CO_2 and activated charcoal to absorb odors. A wick water separator is employed to remove excess water vapor from the O_2 flow.

This sub-function requires the following equipment:

Debris trap
CO_2 & odor absorber
Wick-water separator
Separator pump

Shirt-Sleeve Environment - The Environmental Control System provides a "shirtsleeve" environment during Earth Parking Orbit, Translunar & Transearth Coast, and Lunar Orbit. This function includes the preceding (Circulation of Oxygen; Oxygen Flow Temperature Control, Purification of C/M Atmosphere and Oxygen Flow) as well as the following subfunctions:

[redacted]

Temperature Control of the C/M & S/M Thermal Load

The water-glycol subsystem serves as a heat sink for the C/M and S/M thermal loads which consist of the electronic equipment contained in the spacecraft. The water-glycol subsystem also maintains a constant temperature for a small portion of electronics equipment contained in the C/M thermal load. The equipment used to accomplish this sub-function include:

Glycol evaporator
Cabin heat exchanger
Glycol-to-suit air heat exchanger
Glycol pumps

Cabin Temperature and Pressure Maintenance

During "shirtsleeve" operations, control of the C/M temperature is provided by the cabin temperature control. The control compares the temperature selected by the crew on the cabin temperature selector to that sensed at the cabin heat exchanger inlet by the cabin temperature sensor. Any difference between the selected and sensed cabin temperatures causes the controller to reposition the cabin heat exchanger temperature control valves in such a manner as to reduce the temperature difference to zero. The heating or cooling produced in the cabin is due to the heat rejected to or absorbed from the air in the cabin heat exchanger by the water-glycol. In order to cool the

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cabin, the water glycol inlet temperature to the cabin heat exchanger must be below cabin temperature. Similarly, to heat the cabin the water-glycol inlet temperature to the cabin heat exchanger must be above cabin temperature.

When maximum cooling is required, the cabin heat exchanger temperature control valves direct the entire water-glycol flow first through the cabin heat exchanger where command module heat is absorbed and then through the C/M thermal load, where heat from the electronics equipment is absorbed.

When maximum heating is required the cabin heat exchanger temperature control valves direct the entire water-glycol flow first through the C/M thermal load, where heat is absorbed, and then through the cabin heat exchanger, where heat is rejected to the C/M atmosphere.

C/M pressurization will be provided by the oxygen supply subsystem. During pressure suit or shirtsleeve operation the crew will be under a nominal pressure of 5 psia.

The cabin outflow pressure regulator and negative relief valve has two functions: 1) when the C/M pressure is higher than the external ambient pressure, it limits the differential pressure between C/M and external ambient 2) when the C/M pressure is lower than the external ambient pressure, it limits the differential pressure between external ambient and C/M.

Equipment utilized for this sub-function includes:

- Glycol evaporator
- Glycol-to-suit air heat exchanger
- Cabin heat exchanger
- Space radiators
- Cabin pressure regulator and negative relief valve

<u>Air Circulation of Command Module</u>

Command Module air circulation is provided by two recirculating blowers. A blower selector switch allows the crew to manually select operation of either or both blowers. The cabin air is recirculated by the blower through the cabin heat exchanger where the recirculating cabin air is heated or cooled to maintain the desired cabin temperature.

Post-landing ventilation provides the crew with fresh air from outside the C/M. The equipment consists of two manual shutoff valves - one an inflow valve and the other an outflow valve.

After the C/M has landed the crew will open the inflow & outflow valves and position the cabin recirculating diverter valve for operation of one blower only. This blower pulls ambient air through the inflow valve and pushes it into the cabin where it circulates, providing the crew with fresh air ventilation. This air is then exhausted overboard through the outflow valve.

The equipment used for this sub-function includes the following:

Cabin recirculating blowers.
Inflow & outflow manual control valves
Cabin air shutoff valve

Provision of Potable and Waste Water

During the Apollo mission the fuel cells are the on-board source of potable water. This water is supplied from the service module where the fuel cells are located, to the potable water tank in the command module. Hot water can be obtained by the crew from the hot water supply valve. A portion of the hot potable water from the fuel cells is cooled in the water chiller and cold water can be obtained from the cold water supply valve. Waste water from the pressure suit circuit is reclaimed in the water separator and is channeled into the waste water tank. A water check valve is located in the potable tank discharge line to prevent waste water from contaminating the potable water supply. The waste water & potable water tanks are both pressurized by oxygen.

Water is supplied to the C/M from the fuel cells during all phases of the mission except prior to entry when the S/M is jettisoned.

The equipment used for this sub-function includes the following:

Potable water tank
Waste water tank
Water chiller
Wick-water separator
Oxygen supply

Waste Management

The waste management subsystem (WMS) collects and stores all human waste. The WMS provides bacteria control of urine and a means of jettisoning the urine overboard. The WMS also provides a vacuum cleaner for the collection of solid or water particles in the C/M atmosphere. In addition to the WMS, the storage compartment vent subsystem provides ventilation for the waste, personal hygiene, and food storage compartments.

This subsystem requires the following equipment:

Selector valve
Backup valve
Urine separator
Germicide tank
Vacuum cleaner
Blower
Urine and fecal receptacles

CREW EQUIPMENT SYSTEM

The Crew Equipment System supplements the Environmental Control System and provides the personal equipment required for individual crew needs and comfort.

The Crew Equipment System consists of the following subsystems and their respective major components.

Crew Couches

- Pilot's couch - (fixed)
- Navigator's couch - (removable)
- Systems manager couch - (fixed)

Food Management

- Food
- Food preparation equipment
- Food storage compartment

Waste Management

- Liquid Waste Equipment
- Solid Waste Equipment
- Storage Facilities

Hygiene & Health

Protective Clothing & Accessories

Survival Equipment (after earth touchdown - 3 individual kits)

The space crew and the Crew Equipment System perform the following pertinent functions during a normal lunar landing mission:

Crew Support, Restraint and Protection

This function is accomplished by the three couches, which provides restraint and comfortable support during all mission phases. Each couch will have an adjustable headrest, backrest, hip and knee angle, and arm rest. The couches are designed to accommodate a crewman in a pressure suit or in a shirt sleeve condition. The center couch may be repositioned for use as a sleeping area on the floor beneath the commander's couch and accommodates one crew member.

Included as part of the couch equipment are webbing restraint belts which attach to the pressure suit at the shoulders and hips and also fasten to attach points on the couch. The restraint equipment provides restraint for the crew during launch, entry, weightlessness, and power and control phases.

In addition to the couch and restraints, crew protection is provided by pressure suits, protective clothing and certain accessories. A back pack is also provided for use with the pressure suit in extra-vehicular exploration and maintenance on the moon's surface.

The clothing which provides crew protection and comfort during flight includes a constant wear garment and overwear garment. The pressure suit is worn as required over the constant wear garment.which is worn at all times. When worn by all crew members, the pressure suit constitutes a pressure suit environment. The overwear garment provides the crewman with added protection against radiation and meteorites and is worn as required over the pressure suit.

Hygiene and Health Function and Waste Management

The hygiene and health subsystem provides the crew with a capability to perform the following:

a) Perform bio-medical and physiological monitoring utilizing appropriate equipment.

b) Perform medical treatment when necessary using medical facilities.

c) Adhere to a standard of personal hygiene which applies to body cleaning, oral hygiene, shaving deodorization, and clothing change.

Waste management equipment provides for the sanitary collection, storage and disposal of human waste.

Food Management

Equipment for this purpose provides for the storage, preparation and heating of food as well as for the cleaning of equipment and disposal of waste food bags.

Individual Oxygen Supplies

Oxygen for use in extra-vehicular activity and in the event of evaporation of the Command Module atmosphere is provided by pressure suit back packs.

Crew Survival after Earth Landing

Survival equipment for crew use subsequent to earth landing is provided by NASA.

IN-FLIGHT TEST SYSTEM

The In-Flight Test System (IFTS) is incorporated as a means of improving overall mission reliability. The IFTS provides the crew with an on-board systems check-out and trouble shooting capability with both automatic and manual monitor and test facilities.

The In-Flight Test System consists of the following subsystems:

Crew Control Panel

Display Lights

Programmer

Reference Voltage Supply

Reference Voltage Selector Gates

Comparator

Storage Register

. Stimuli Generator

. Test Point Panel

. Manual Test Unit

The space crew and the In-Flight Test System perform the following pertinent functions during a normal lunar landing mission:

<u>Automatic Systems Checkout</u>

The IFTS, in conjunction with the main console displays, provides the following automatic test capabilities:

Pre-operational readiness check

Conditioning monitoring during operation

Indications of malfunctions or unsafe conditions

Performing a self-test cycle

The automatic IFTS which is operated upon command from the crew, scans approximately 200 test points for out of tolerance conditions. The existence of such a condition is indicated by a no-go light on the main console display panel and an alpha-numeric visual readout on the IFTS display lights. The alpha-numeric readout identifies the test-point by system and location on the IFTS test point panel.

Manual Systems Checkout

The manual test unit consisting of a cathode ray tube oscilloscope and volt-ohm meter provides the crew with additional test capability (trouble shooting). Normalized voltage analogs of each of the approximately 200 test points are available on the test point panel. The manual test unit may also be used to measure any other accessible test points in the space craft. Stimulus - response testing may also be performed either manually or automatically.

The following represent additional system capabilities which may be realized by combining the decision making capability of the crew with the monitoring capability of the IFTS:

Detection of out-of-tolerance conditions;

Isolation of an out-of-tolerance condition to a modular or component level;

Determination of the criticality of the out-of-tolerance condition; and

Verification of any corrective action which may be required.

ELECTRICAL POWER SYSTEM

The Electrical Power System provides all the electrical power which is necessary to complete the lunar landing mission. It also provides electrical power for any abort maneuver.

The Electrical Power System consists of the following subsystems:

Fuel Cells
Entry Batteries
Post Landing Battery
Battery Chargers
Inverters
Wiring & Busses
Controls & Displays

The space crew and the Electrical Power System perform the following pertinent functions during a normal lunar landing mission:

Main Power (AC & DC) - The Electrical Power System supplies AC and DC electrical power to the Apollo spacecraft from lift-off to jettisoning of the Service Module. This function requires fuel cells, inverters, wiring and busses, and controls and displays.

Entry Power (AC & DC) - Subsequent to jettisoning of the Service Module, the Electrical Power System supplied AC and DC electrical power to the Command Module during entry into the earth's atmosphere and parachute descent to earth touchdown. This function requires entry batteries, inverters, wiring and busses, and controls and displays.

Overload Power (AC & DC) - In the event of a contingency, the Electrical Power Supply supplies AC and DC electrical power for any circuit overload situation that might arise. This function requires entry batteries, fuel cells, inverters, wiring and busses, and controls and displays.

[redacted]

Post Landing Power (AC & DC) - Following earth touchdown, the Electrical Power System supplies AC & DC electrical power to the Command Module for all post-landing operations. This function requires post landing battery, inverters, wiring and displays.

Entry Battery Recharging - The Electrical Power System provides recharging of the entry batteries after usage for overload conditions or prior to usage for entry power supply. This function requires battery chargers, entry batteries, fuel cells, inverters, wiring & busses, and controls and displays.

Post-Landing Battery Recharging - The Electrical Power System provides recharging of the post-landing battery just prior to Service Module jettison. This function requires battery chargers, post-landing battery, fuel cells, inverters, wiring & busses, and controls and displays.

LAUNCH ESCAPE SYSTEM

The Launch Escape System provides the propulsion capability to separate the Command Module from the Service Module and Launch Vehicle in the event of any contingency which results in a pad or boost abort. The Launch Escape System provides this capability up to the time of its normal programmed jettison approximately 5 seconds after S-II engine ignition.

The Launch Escape System consists of the following subsystems:

Launch Escape Tower

Launch Escape Motor

Tower Jettison Motor

Pitch Control Motor

Launch Escape Sequence Controller

The space crew and the Launch Escape System perform the following pertinent functions during a normal lunar landing mission:

Abort Capability - For abort prior to lift-off or during boost until approximately 5 seconds after S-II engine ignition, the Launch Escape System provides an abort capability. In the event of an abort, the Launch Escape Motor propels the Command Module to a safe distance from the Service Module and Launch Vehicle. A few seconds after thrust from the Launch Escape Motor is terminated, the Tower jettison motor propels the Launch Escape System away from the Command Module allowing safe deployment of the Earth Landing System. The Pitch Control Motor provides thrust to pitch the flight attitude of the Launch Escape System away from the Command Module flight path. The Launch Escape System Sequence Controller, located in the Command Module, initiates activation signals for operation of the Launch Escape System. A manual control override capability to activate the Launch Escape System.

is provided for both spacecraft crew and ground control operation.

Normal Jettison - During normal flight, the Launch Escape System is jettisoned approximately 5 seconds after S-II engine ignition. The Tower Jettison Motor propels the Launch Escape System away from the Command Module & Launch Vehicle. The Pitch Control Motor provides thrust to pitch the flight attitude of the Launch Escape System away from the Command Module & Launch Vehicle flight path. The Launch Escape System Sequences Controller initiates activation signals for operation of the Launch Escape System is provided for both spacecraft crew and ground control operation.

EARTH LANDING SYSTEM

The Earth Landing System returns the Command Module safely to earth after normal entry from a lunar landing mission. It also safely returns the Command Module following any abort maneuver.

The Earth Landing System consists of the following subsystems:

- Drogue Parachute
- Landing Parachute Subsystem
- Impact Attenuation Subsystem
- Landing Location Aids

The space crew and the Earth Landing System perform the following pertinent functions during a normal lunar landing mission:

Spacecraft Stabilization - Following entry of the Command Module into the earth's atmosphere, the Earth Landing System stabilizes the Command Module. This is accomplished during early descent by the Drogue parachute.

Velocity Control - The Earth Landing System reduces velocity during descent through the use of the landing parachutes.

Impact Attenuation - The Earth Landing System reduces touchdown velocity such that that the Command Module structure is not impaired. This function requires use of the C/M Heat Shield and the Impact Attenuation Subsystem.

Recovery Aids - The Earth Landing System provides location and survival aids necessary for safe and prompt recovery of the spacecraft and crew.

[redacted]

COMMAND MODULE STRUCTURAL & HEAT PROTECTION SYSTEM

The Command Module Structural and Heat Protection System carries all structural loads, houses the 3 crew members, and provides mounting for the Command Module Systems.

The Command Module Structural and Heat Protection System consists of the following subsystems.

Crew Compartment

Aft Compartment

Forward Compartment

Heat Shield

Earth Impact Attenuation

The Command Module Structural and Heat Protection System performs the following pertinent functions:

Mounting Support - The C/M Structural and Heat Protection System provides a mounting surface to support all Command Module systems.

Pressurization - The C/M Structural and Heat Protection System provides a vessel which can be pressurized to protect the crew and spacecraft systems.

Thermal Protection - The C/M Structural and Heat Protection System provides thermal protection during the maximum heating of entry into the earth's atmosphere.

Radiation Protection - The C/M Structural and Heat Protection System has the capability to decrease the flux density due to nuclear radiation.

[redacted]

Meteoroid Protection - The C/M Structural and Heat Protection System provides protection against the damaging effects of meteoroids.

Impact Attenuation - The C/M Structural and Heat Protection System provides attenuation of the loads imposed by earth landing impact.

Load Support - The C/M Structural and Heat Protection System carries all ground and flight loads for a normal mission or any abort maneuver.

Visual Capability - The C/M Structural and Heat Protection System provides visual capability to the crew during the mission.

SERVICE MODULE STRUCTURAL SYSTEM

The Service Module Structural System provides mounting for all Service Module systems from lift-off to jettisoning of the Service Module.

The Service Module Structural System consists of the following subsystems:

Engine Compartment

Equipment Compartment

Antenna Doors

ECS Radiations

Meteoroid Protection

The Service Structural System performs the following pertinent functions:

Mounting Support - The S/M Structural System provides a mounting surface to support all Service Module systems.

Meteoroid Protection - The S/M Structural System provides protection for equipment against the damaging effects of meteoroids.

Radiation Protection - The S/M Structural System has the capability to decrease the flux density due to nuclear radiation.

Load Support - The S/M Structural System carries all ground & flight loads for a normal mission or any abort maneuver.

CONTROLS AND DISPLAYS SYSTEM

The Controls and Displays System provides a vital interface between the three-man crew and the spacecraft systems, and enables the crew to monitor and control all system activity during the mission. With the exception of the Ascent, Earth Parking Orbit, and Translunar Injection Phases in which the crew monitors flight parameters and spacecraft systems operation, the control of the Apollo spacecraft is essentially manual, i.e., all semi-automatic functions are initiated by a crewman. The crew receives various types of information from the displays panel allowing them to make necessary decisions regarding the many tasks required during the mission. The spacecraft controls are utilized by the crew in implementing the necessary action based on these decisions. The panel displays also allow the monitoring of all systems for rapid detection of out-of-tolerance conditions. In addition to the manual initiation of semi-automatic functions, a manual override and control capability of these functions is provided to the crew.

The Controls and Displays System consists of the following items (particular controls and/or displays) and their functions. Figure 60 shows the Apollo Display and Control Panel.

Control or Display	Function
Gimbal Position Indicator	Displays the angular position of the service module engine with respect to the spacecraft x-axis. Two controller knobs allow the astronaut fo adjust the gimbal position.
Δ V Display	Monitor and control of velocity corrections. Displays Δ V remaining. Magnitudes of anticipated velocity corrections range from 10 ft/sec to 9990 ft/second.
SCS Control Panel (Stabilization and Control System)	Push-button selection of mode operation. Dead band adjustment may be set for 0.5°, 5.0°, or open, on each of the three attitude control channels. Push buttons allow any channel to be disabled for trouble-shooting.
Flight Director Attitude Indicator	Attitude is shown by a gimballed ball driven by the guidance system. Three attitude rate indicators and three attitude error indicators are also provided.
Rotational and Translational Controllers	The left and center seat positions have controllers. A thumb-actuated stick provides attitude control. Stick provides attitude control. Stick forward causes pitch rate downward. Stick backward causes pitch rate upward. Stick left and right cause roll rate left and right respectively. Rotation of the grip causes yaw. A similar stick provides translational control.

Control or Display	Function
Projection Viewer	Displays tables of data stored on microfilm.
Computer Keyboard and Readout	Manual selection of computer program. Has twelve keys for digits 0 through 9 and + and - to insert data. Several registers and readouts are displayed. Also, computer condition is displayed.
Clock: Greenwich Mean Time (Dial)	This is a dial clock with hands for hours, minutes and seconds. Days are displayed on a two digit readout.
Clock: Greenwich Mean Time (Digital)	Same as the dial clock except that the display is digital.
Clock: Time From Event	Digital clock displaying time from a pre-selected event.
Clock: Time To Event	Digital clock displaying time to a pre-selected event.
Audio Control Panel	Listening level is controlled by a thumbwheel audio volume control. Four switches select mode of transmission. They are HF, VHF/AM, DSIF, and INTERCOM. Each switch has the three positions T/R, OFF and REC. T/R, permits two-way communication. REC permits receive only. The Audio Control Panel also allows the selection of PTT (Push To Talk) or VOX (Voice Operated Relay) modes. There is a thumbwheel to select VOX threshold.

Control or Display	Function
Entry Monitoring Indicator	A moving marker displays G-loading as a function of time, starting with the commencement of entry. Two lines drawn on the display show the permissible upper and lower limits of G-loading as a function of time. If the marker crosses the upper line, skipout is implied. Crossing the lower line implies excessive G-loading, or excessive entry heating.
S/M Quad. Temp.	Push buttons select the service module quadrant whose temperature is to be displayed. The display is by meter.
RCS	Four interlocked bushbuttons and a rocker switch select the RCS system to be displayed and/or controlled. The displayed parameters are: helium tank pressure, helium regulator pressure, package temperature, fuel quantity, and oxidizer quantity. Control functions are individual propellant shutoff and C/M pressurization controls. Propellant shutoff switches are used in conjunction with the pushbutton and rocker switch system selectors.
Lighting	The lighting controls are duplicated on the left and right sides of the capsule. The primary set of flood lights is turned on and off and adjusted for brightness by a thumbwheel. The secondary set of flood lights is controlled by an on-off switch only. A push button operates all annunciator (warning) lights and devices to test them.

Control or Display	Function
Barometer	A dial-face barometer provides altitude information while in the atmosphere.
Booster Situation (Event-Time) Indicator	Passage of major events such as ignitions, cutoffs, and separations is indicated by a line of lights. The sequence runs from S-I IGN through S-IV SEP.
Antenna Control	The Omni (omnidirectional) or dish antenna is selected by a two-position switch. The dish antenna may be set for automatic or manual control deployed or retracted. Two 2-position switches slew the dish left or right and up or down, while two meters show actual antenna position. Another meter shows ACG level.
Master Caution Lights	A block of lights indicates a system malfunction. Detail of information given is system level only.
IFTS (In-Flight Test System)	The in-flight test system is operated by a switch with the positions SCAN and OFF. In the SCAN position, a large number of test points are sampled. NO-GO conditions are displayed on the IFTS panel, which is off the main control panel. The display is alpha-numeric and indicates test point and system.

Control or Display	Function
Fuel Cells	The Fuel Cells panel allows control and monitoring of three H_2-O_2 fuel cells. A flow rate meter displays H_2 or O_2 flow rate, selected by a two-position switch and a system indication four-position switch. The pH of the reaction product, water, is shown on a meter. This reading may be selected for any fuel cell independently, or for all collectively. A meter displays temperature, selected from radiator exit (coolant) fuel, cell skin, or condenser exhaust in the selected fuel cell. Also the pressure of O_2, H_2, or N_2 may be selected and read. There are Purge, Start, Stop switches for H_2 and O_2. These initiate an increased flow of gas to clean the electrodes in the selected fuel cell. Any fuel cell may be shut down by pushing a shutoff command button while holding down a button referring to the appropriate fuel cell.
Power Distribution Display No. 1.	Six 2-position switches connect any of fuel cells 1, 2, or 3 to either buss A or B.
Power Distribution Display No. 2.	A voltmeter indicates voltage of Buss A or B, battery charger, post landing buss, or battery A, B, or C, as selected by pushbutton. Similarly, current may be read on fuel cell 1, 2, or 3, battery A, B, or C, or battery charger. A warning light goes on if the DC voltage drops below 25 V, and stays on until the reset button is pushed. (Continued)

Control or Display	Function
Power Distribution Display No. 2 (Continued)	Switches are used to connect battery A or B to the DC busses.
Power Distribution Display No. 3	Meters display AC Volts and Frequency. Pushbuttons select phase A, B, or C; buss 1 or 2. A meter displays the component temperature, selected by pushbutton, for Inverter 1, 2, or 3, battery A, B, or C, and the sequencer. (The sequencer is a control circuit for separation system). Pushbuttons control battery charger input power and select battery (A, B, or C) to be charged. Two-position switches are used for the following: DC input to Inverters 1, 2, and 3; and connection of Inverters 1, 2, or 3 to AC buss 1 or 2. Also, there are failure (voltage low) warning lights for each AC buss group.
Telecommunications	<u>VHF AM</u>: A mode switch has positions - RECEIVE, OFF, STANDBY where the standby position is for transmission of voice only. A two-position switch selects the frequency of reception. The TV camera is controlled by an ON-OFF switch. A RADIO RELAY switch with the positions LOCAL and EARTH LINK is set at EARTH LINK to relay transmission from the LEM to Earth.

Control or Display	Function
Telecommunications (Continued)	There is an ON-OFF switch for the C-Band transmitter which is used for tracking.

There is an ON-OFF switch for VHF FM which is used for telemetry. A tape recorder may be turned ON or OFF, set for RECORD or PLAY, or for FORWARD, REVERSE, FAST FWD or FAST REVERSE.

A VHF recovery beacon may be set for ON, OFF, or AUTO. In the AUTO position the beacon goes on automatically when the parachutes open. Also the mode of transmission may be selected among CW, VOICE, OFF and BCN, which is an amplitude modulated tone.

There is a provision for POWER selection by push buttons for OFF, 200 MW, 5 W, 20W, and STANDBY, where STANDBY supplies power to heat the filaments only.

The oscillator has three modes of operation, set by push buttons:

AUX The oscillator is crystal controlled.

VCO The oscillator is controlled by a radio signal from earth.

Control or Display	Function
Telecommunications (Continued)	NORM The oscillator is under Earth control unless the Earth signal becomes too weak, in which case the oscillator is automatically switched to crystal control.
	One of three PCM (pulse code modulation) FORMATS may be selected for telemetry transmission. This provides the option of using less power at the expense of decreased transmission rate.
	The MODE of transmission may be selected among: TLM/VOICE, RANGING, NARROW BAND TLM, TAPE PLAY/VOICE, TV, TLM, VOICE, and KEY.
ECS (Gas) Display	The atmospheric pressure and temperature for both the cabin interior and the suit inlet are displayed. CO_2 partial pressure in the suit circuit is displayed. A two-position switch selects which compressor is used. Cabin blowers 1 and 2 each have an ON-OFF switch. Two thumb wheels set the thermostats for cabin and suit temperature.
ECS (Liquid) Display	ON-OFF switches route coolant through any combination of four space radiators. Coolant pump 1 or 2 may be selected by means of a two-position switch. Coolant inlet and outlet temperatures are

Control or Display	Function
ECS (Liquid) Display (Continued)	displayed, where inlet means out from the radiators. Discharge pressure of the glycol coolant after pumping is displayed, and also the reservoir quantity. The reservoir supplies coolant to make up for leakage in the cooling system. The quantity of potable water is displayed. A supplementary cooling mode is provided, that being the evaporation of water into the vacuum of space. The temperature of the steam from this evaporator is displayed.
Cryogenic Display	Pressure and temperature are displayed for two tanks each of H_2 and O_2. A two-position switch is used to select H_2 or O_2 indication. Any tank may be isolated by setting at CLOSED an appropriate pair of ISOLATE and SHUTOFF switches to connect either O_2 tank to the ECS system or to the fuel cells.
Service Propulsion	Two meters display quantity of fuel and oxidizer, and a three-position switch selects Tank A, Tank B, or total for display. The ratio of propellants being fed to the engine is displayed, and there is a switch to select lean or rich mixture. The mixture is adjusted so that fuel and oxidizer are exhausted at a ratio of two parts oxidizer to one part fuel. A light warns of high chamber wall

Control or Display	Function
Service Propulsion (Continued)	temperature. Two dual meters display the pressures of tank and inlet fuel and oxidizer. Also, helium tank pressure and temperature are displayed. Two H_2 regulators are controlled by normally closed On- Off switches. Both switches are opened prior to engine firing. Four event lights indicate when pairs of engine injector valves are opened by SCS prior to engine firing.
Launch Escape	A switch with the positions ARM and SAFE is used to arm the Launch Escape System. A light indicates READY when the system is armed. Two other lights indicate TOWER RELEASED or TOWER NOT RELEASED. Three buttons located on the pilot's arm rest may be used to start the sequencer for abort; start the launch escape motor in case the sequencer fails; and release the launch escape tower.
SCS Power Control	An ON-OFF switch supplies power to the stabilization and control system. The system may be shut down for trouble shooting. Portions of the SCS system may be shut down individually.
Abort Lights	There are three abort lights on the main panel and one in the lower equipment bay. They indicate that an abort mode has been entered, whether astronaut or GOSS initiated.

Control or Display	Function
Separation System	An ARM-SAFE switch arms the separation system. A READY light indicates that the system is armed. Push buttons activate service module posigrade acceleration, and retrograde for the S-IV booster, the service module and the LEM. Also there are buttons to separate the command module from the service module, the LEM from the spacecraft, the booster from the spacecraft, the adapter from the spacecraft, and to jettison the shroud that covers the service module engine.
Circuit Breakers	The circuit breakers are located on two side panels, to the left side of the pilot and to the right side of the systems engineer.

[redacted]

The following displays and controls are located in the lower equipment bay:

Film Strip Viewer	There is a film strip viewer in the lower equipment bay.
Clocks	Greenwich Mean Time and time to event are displayed.
IFTS patchboard	A pattern of lights gives an alpha-numeric readout of out-of-tolerance test points. There is a patchboard of electrical sockets allowing access to the test points for trouble shooting. An oscilloscope and VOM are also provided.
Sextant and Scanning Telescope	A hand controller with two degrees of freedom allows slewing of the shaft angle and trunnion angle. The speed of slewing may be set by a switch for high, low, or medium. There are hand cranks as a manual backup to move the scanning telescope. By slaving the sextant to the telescope, the sextant may be moved too. There is a digital display of telescope shaft and trunnion angle. OPTICS CONT MODE switch with the positions DIRECT and RESOLVED allows the option of having the two degrees of

[redacted]

freedom of the OPTICS HAND CONTROL be in polar coordinates (DIRECT) or in rectangular coordinates (RESOLVED). There is an ON-OFF sextant power switch. When the shaft and trunnion angles are set so that a star and landmark are brought into coincidence in the sextant, the crew punches a MARK button, which enters the trunnion angle and shaft angle into the computer. A three position OPTICS MODE switch has the positions ZERO SHAFT LOCK, which allows trunion motion only. A SCT SLAVE switch has the positions STAR LOS (line of sight) which causes the scanning telescope to follow the sextant; OFFSET 25° which moves the telescope trunnion angle by +25°; and TRUNNION 0°, which sets the telescope trunion at 0°.

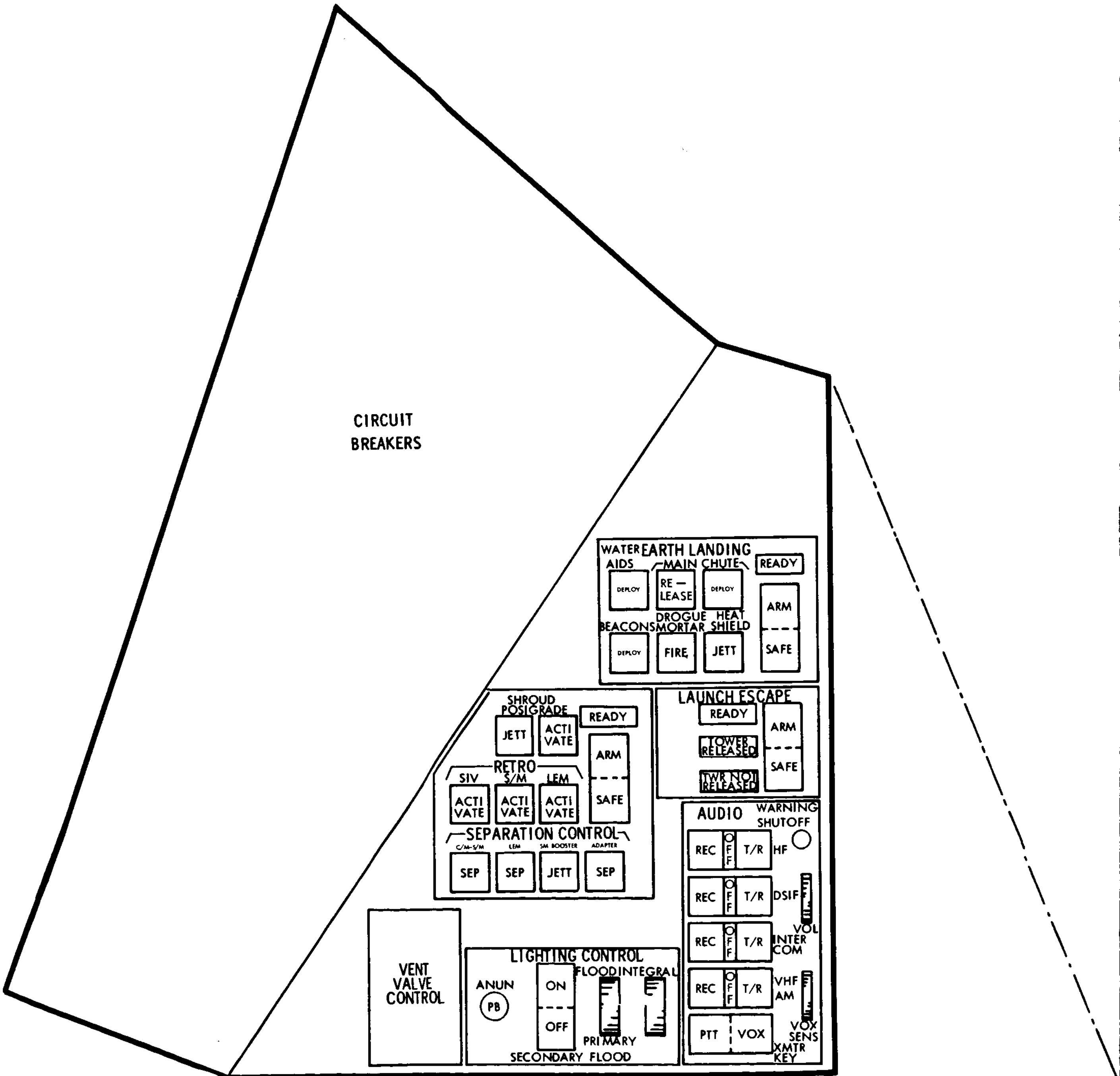
Coupling Display Unit (CDU)

CDU shaft angles for roll, pitch, yaw and sextant shaft and trunnion are displayed digitally, and may be set by slew switches or by manual thumb wheels. Two push buttons allow selection of manual or automatic input to the CDU. There are six CDU MODE CONTROL push buttons, marked ZERO ENCODER, COURSE

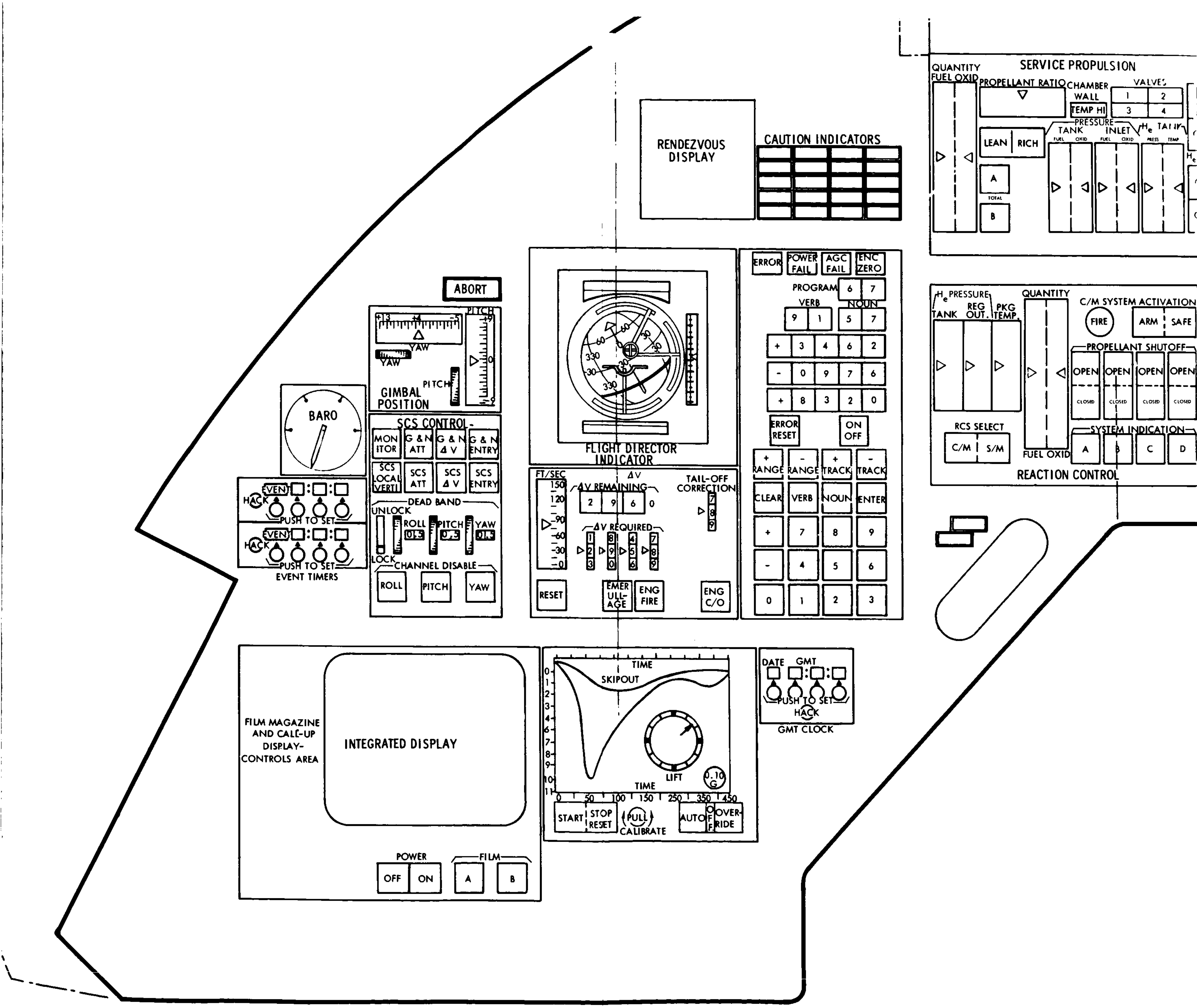
Inertial Measurements Unit	Two windows in the sextant control panel allow visual inspection of the inertial measurements unit (IMU). There is an IMU TEMP MODE which allows selection of the mode of temperature control for the IMU. The normal positions of this switch is AUTO OVRD.
Attitude Error Indicator	There is an attitude error indicator identical to that on the main panel.
Computer	There is a computer keyboard and display identical to that on the main panel.
3 Axis Control	The rotational hand-controller may be connected in the lower equipment bay.
Cabin lights	An intensity control for cabin lights is provided in the lower equipment bay.

The Main Display panel is immediately in front of the 3 man crew when they are in the crew couches, i.e., the panel is perpendicular to the x-axis of the spacecraft. The crewmen are located relative to the display panel as follows:

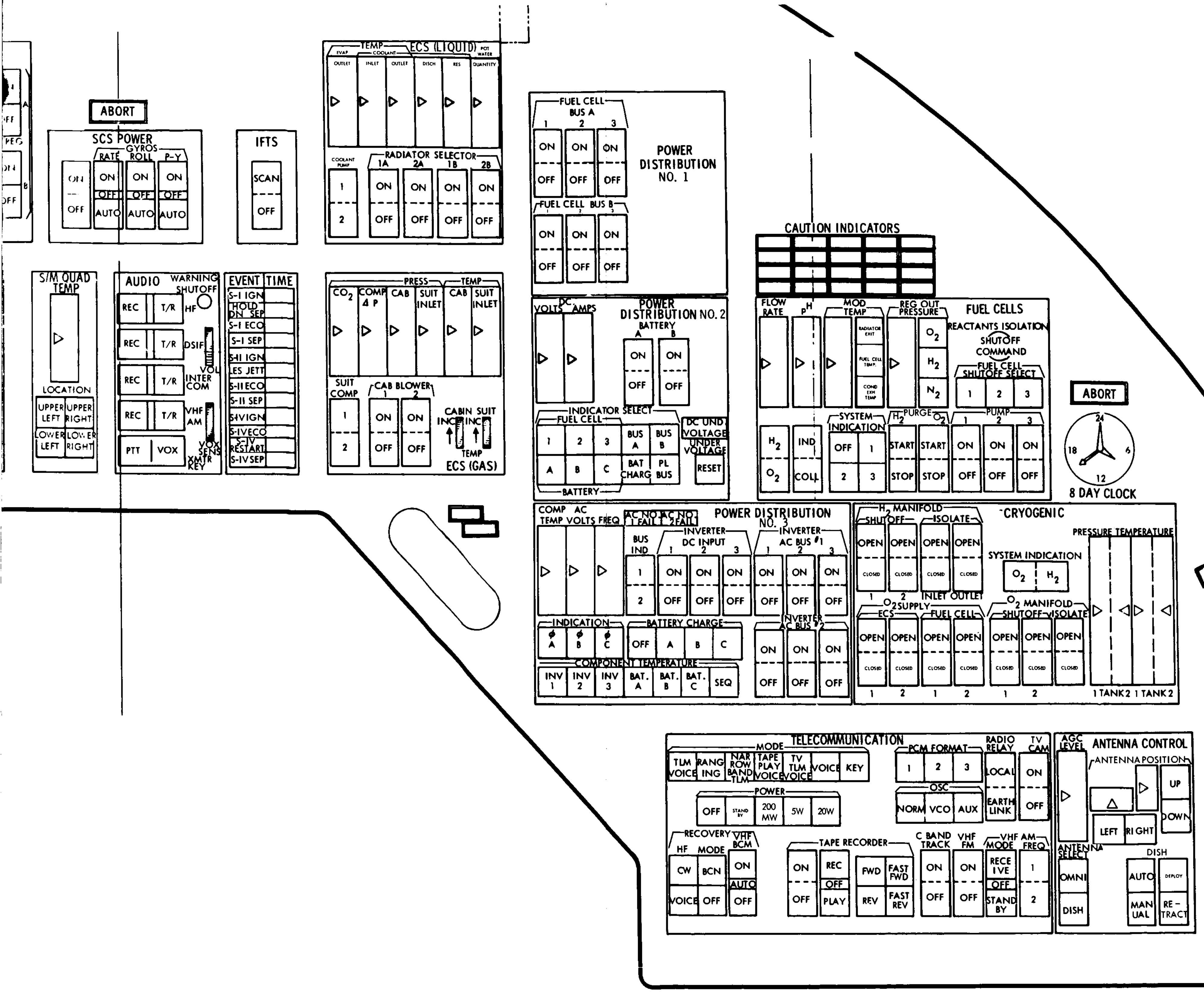
Crew	Location & Function
a) Pilot	The pilot's couch is the left-most position when facing up the x-axis. His primary duty is the flight of the spacecraft - initiating rocket firings, changing the attitude, etc. Also, the pilot is first in command.
b) Navigator	The navigator determines trajectory and velocity changes needed. He is second in command and occupies the center couch.
c) Engineer	The engineer operates the environmental control system and is responsible for fuel management. His station is at the right; and he is third in command.



CIRCUIT
BREAKERS
WATER AIDS
EARTH LANDING
MAIN CHUTE
READY
DEPLOY
RE-LEASE
DEPLOY
ARM
SAFE
DROGUE
HEAT SHIELD
BEACONS
MORTAR
DEPLOY
FIRE
JETT
LAUNCH ESCAPE
READY
ARM
TOWER RELEASED
SAFE
TWR NOT RELEASED
SHROUD
POSIGRADE
READY
JETT
ACTI VATE
ARM
RETRO
SIV
S/M
LEM
ACTI VATE
ACTI VATE
ACTI VATE
SAFE
SEPARATION CONTROL
C/M-S/M
LEM
SM BOOSTER
ADAPTER
SEP
SEP
JETT
SEP
AUDIO
WARNING SHUTOFF
REC
OFF
T/R
HF
REC
OFF
T/R
DSIF
VOL
REC
OFF
T/R
INTER COM
REC
OFF
T/R
VHF AM
PTT
VOX
VOX SENS
XMTR KEY
VENT VALVE CONTROL
LIGHTING CONTROL
ANUN
PB
ON
OFF
FLOOD
INTEGRAL
PRIMARY
SECONDARY FLOOD


SERVICE PROPULSION
QUANTITY
FUEL OXID
PROPELLANT RATIO
CHAMBER WALL TEMP HI
VALVES
PRESSURE
TANK
INLET
FUEL OXID
LEAN
RICH
TOTAL
RENDEZVOUS DISPLAY
CAUTION INDICATORS
ABORT
GIMBAL POSITION
PITCH
YAW
BARO
SCS CONTROL
MON ITOR
G & N ATT
G & N Δ V
G & N ENTRY
SCS LOCAL VERT
SCS ATT
SCS Δ V
SCS ENTRY
DEAD BAND
UNLOCK
LOCK
ROLL
CHANNEL DISABLE
EVENT
HACK
PUSH TO SET
EVENT TIMERS
FLIGHT DIRECTOR INDICATOR
FT/SEC
Δ V
Δ V REMAINING
Δ V REQUIRED
TAIL-OFF CORRECTION
RESET
EMER ULL-AGE
ENG FIRE
ENG C/O
ERROR
POWER FAIL
AGC FAIL
ENC ZERO
PROGRAM
VERB
NOUN
ERROR RESET
ON OFF
RANGE
TRACK
CLEAR
ENTER
He PRESSURE
TANK
REG OUT.
PKG TEMP.
QUANTITY
FUEL OXID
C/M SYSTEM ACTIVATION
FIRE
ARM
SAFE
PROPELLANT SHUTOFF
OPEN
CLOSED
RCS SELECT
C/M
S/M
SYSTEM INDICATION
REACTION CONTROL
FILM MAGAZINE AND CALL-UP DISPLAY-CONTROLS AREA
INTEGRATED DISPLAY
POWER
OFF
ON
FILM
TIME
SKIPOUT
LIFT
START
STOP RESET
PULL
CALIBRATE
AUTO
OFF
OVER-RIDE
DATE
GMT
GMT CLOCK

2


ABORT
SCS POWER
GYROS
RATE
ROLL
P-Y
IFTS
SCAN
ECS (LIQUID)
TEMP
RADIATOR SELECTOR
FUEL CELL BUS A
FUEL CELL BUS B
POWER DISTRIBUTION NO. 1
CAUTION INDICATORS
S/M QUAD TEMP
LOCATION
AUDIO
WARNING SHUTOFF
EVENT TIME
PRESS
SUIT COMP
CAB BLOWER
CABIN TEMP
SUIT TEMP
ECS (GAS)
POWER DISTRIBUTION NO. 2
BATTERY
INDICATOR SELECT
FUEL CELL
DC UNDERVOLTAGE
RESET
FUEL CELLS
REACTANTS ISOLATION SHUTOFF COMMAND
FUEL CELL SHUTOFF SELECT
SYSTEM INDICATION
PURGE
PUMP
8 DAY CLOCK
POWER DISTRIBUTION NO. 3
INVERTER DC INPUT
INVERTER AC BUS #1
INVERTER AC BUS #2
BATTERY CHARGE
COMPONENT TEMPERATURE
CRYOGENIC
H_2 MANIFOLD
O_2 SUPPLY
O_2 MANIFOLD
PRESSURE TEMPERATURE
TELECOMMUNICATION
MODE
PCM FORMAT
RADIO RELAY
TV CAM
POWER
OSC
RECOVERY
TAPE RECORDER
C BAND TRACK
VHF FM
VHF AM
ANTENNA CONTROL
AGC LEVEL
ANTENNA POSITION
ANTENNA SELECT
DISH

3

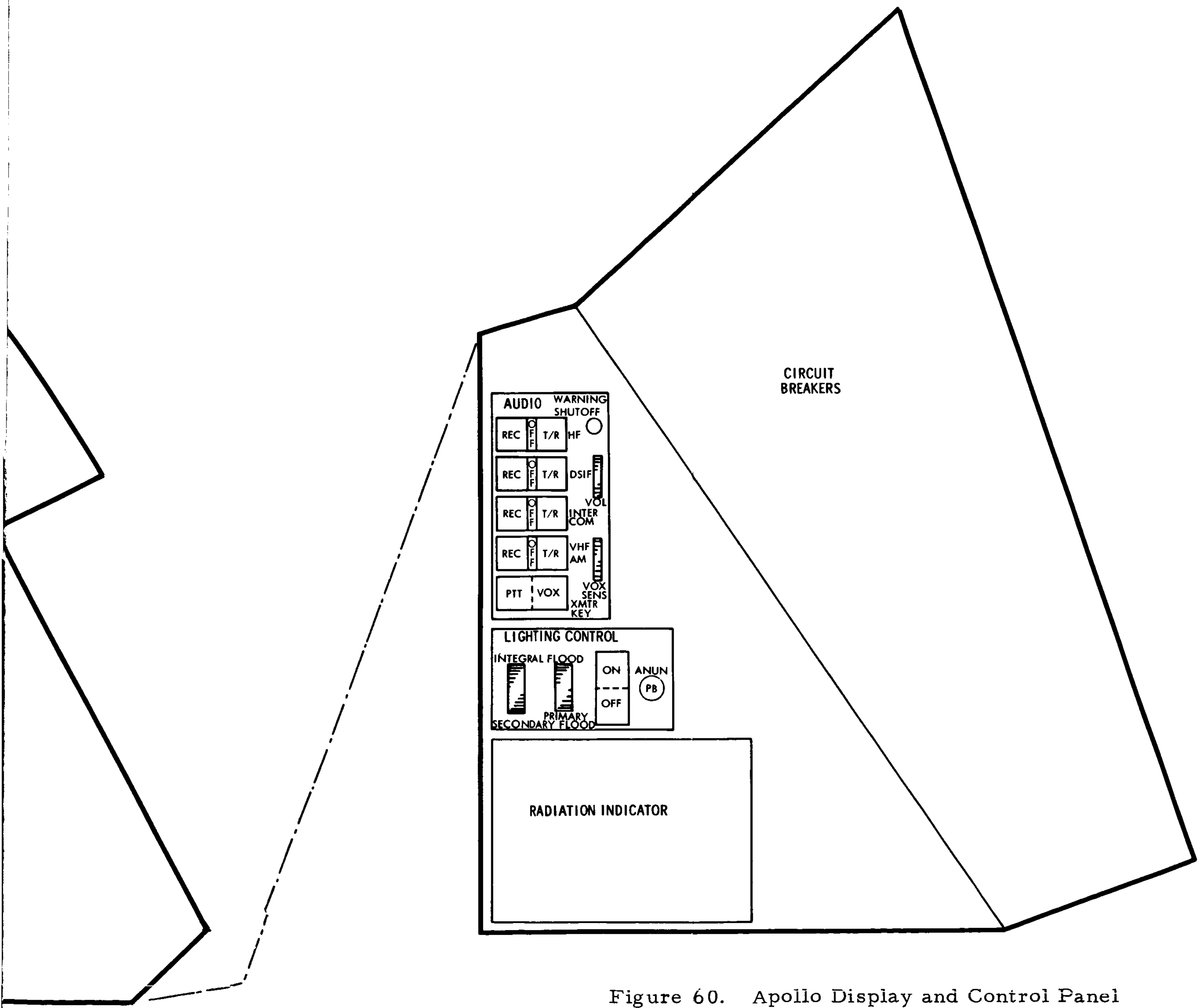


Figure 60. Apollo Display and Control Panel

APPENDIX G

GROUND OPERATIONAL SUPPORT SYSTEM

The Ground Operational Support System (GOSS) is a complex of tracking stations, computational facilities, and control stations which contributes to the probability of mission success and crew safety by providing support to the S/C as required.

The GOSS consists of an Integrated Mission Control Center (IMCC), a Launch Control Center (LCC), one or more Recovery Control Centers (RCC), the augmented Mercury tracking network, and three Deep Space Instrumentation Facilities (DSIF). The coverage provided by this network for the lunar landing mission described in this document is shown graphically in Figure 61 and in tabular form in Table 3.

The operational support consists of the following functions:

(1) Monitoring - During pre-launch and ascent phases, the GOSS will receive, evaluate, and act upon data - some of which is not available to the spacecraft. During other phases of the mission, the GOSS will provide alternative interpretation of critical data on status and performance of on-board instruments and systems.

(2) Navigational Backup - In all phases of a mission, the GOSS will determine the S/C trajectory based upon radar tracking data and provide this information to the crew.

(3) Weather Information - The GOSS will provide information on both solar weather and weather in the vicinity of the landing sites.

(4) <u>Diagnostic</u> - The GOSS will provide a diagnostic capability to assist the crew in isolating failures. The GOSS will recommend appropriate courses of action in these cases.

In addition to such operational support, the GOSS will collect data for post-flight analysis. The GOSS will also handle the public information function.

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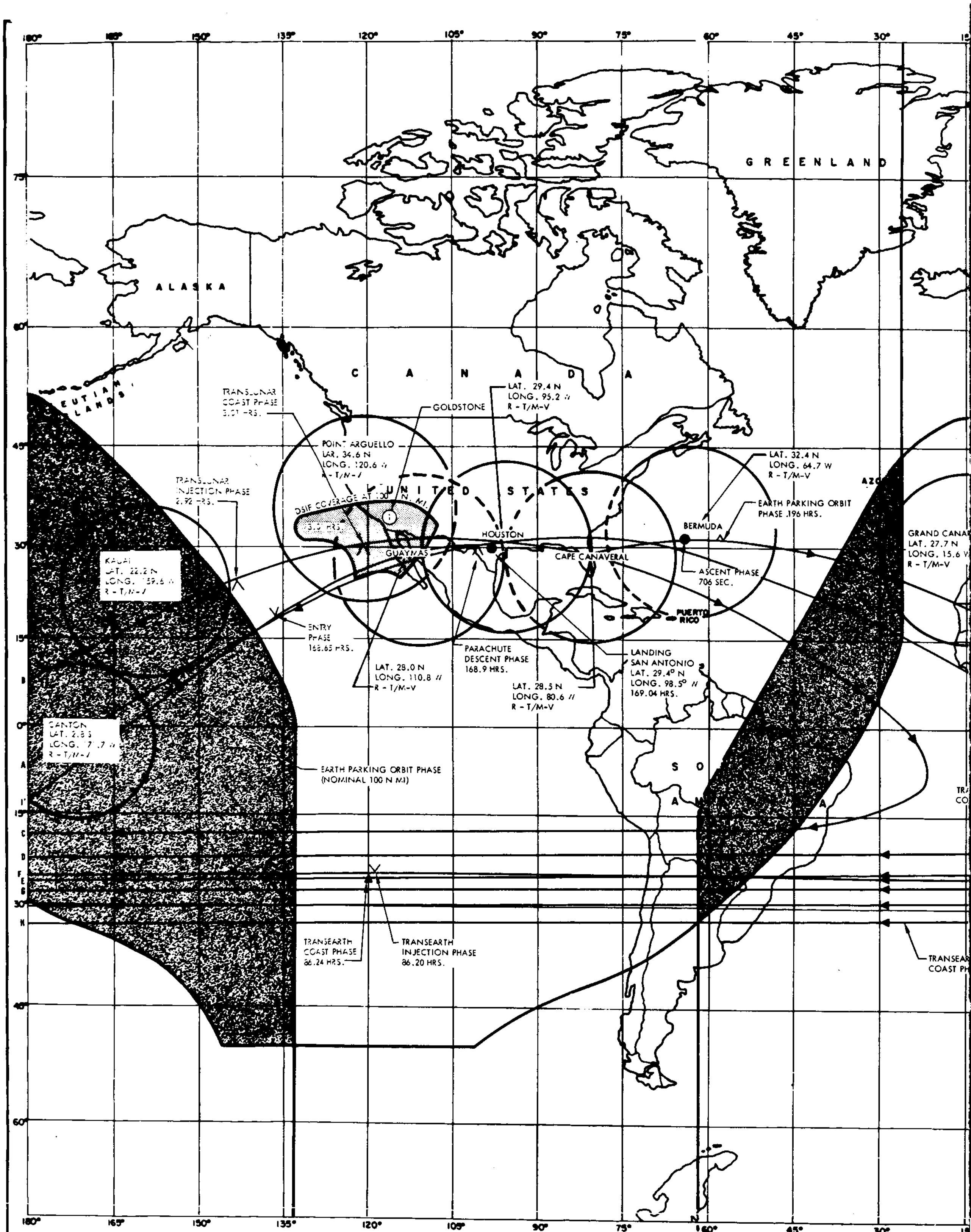

180°
165°
150°
135°
120°
105°
90°
75°
60°
45°
30°
75°
60°
45°
30°
15°
0°
15°
30°
45°
60°
GREENLAND
ALASKA
CANADA
UNITED STATES
TRANSLUNAR COAST PHASE
GOLDSTONE
LAT. 29.4 N
LONG. 95.2 W
R - T/M-V
POINT ARGUELLO
LAT. 34.6 N
LONG. 120.6 W
R - T/M-V
TRANSLUNAR INJECTION PHASE
2.92 HRS.
DSIF COVERAGE AT 100 N. MI.
KAUAI
LAT. 22.2 N
HOUSTON
GUAYMAS
CAPE CANAVERAL
BERMUDA
LAT. 32.4 N
LONG. 64.7 W
R - T/M-V
EARTH PARKING ORBIT PHASE .196 HRS.
ASCENT PHASE
706 SEC.
GRAND CANARY
LAT. 27.7 N
LONG. 15.6 W
R - T/M-V
PUERTO RICO
ENTRY PHASE
168.65 HRS.
PARACHUTE DESCENT PHASE
168.9 HRS.
LANDING
SAN ANTONIO
LAT. 29.4° N
LONG. 98.5° W
169.04 HRS.
LAT. 28.0 N
LONG. 110.8 W
R - T/M-V
LAT. 28.5 N
LONG. 80.6 W
R - T/M-V
CANTON
EARTH PARKING ORBIT PHASE
(NOMINAL 100 N MI)
TRANSEARTH COAST PHASE
86.24 HRS.
TRANSEARTH INJECTION PHASE
86.20 HRS.
TRANSEARTH COAST PHASE

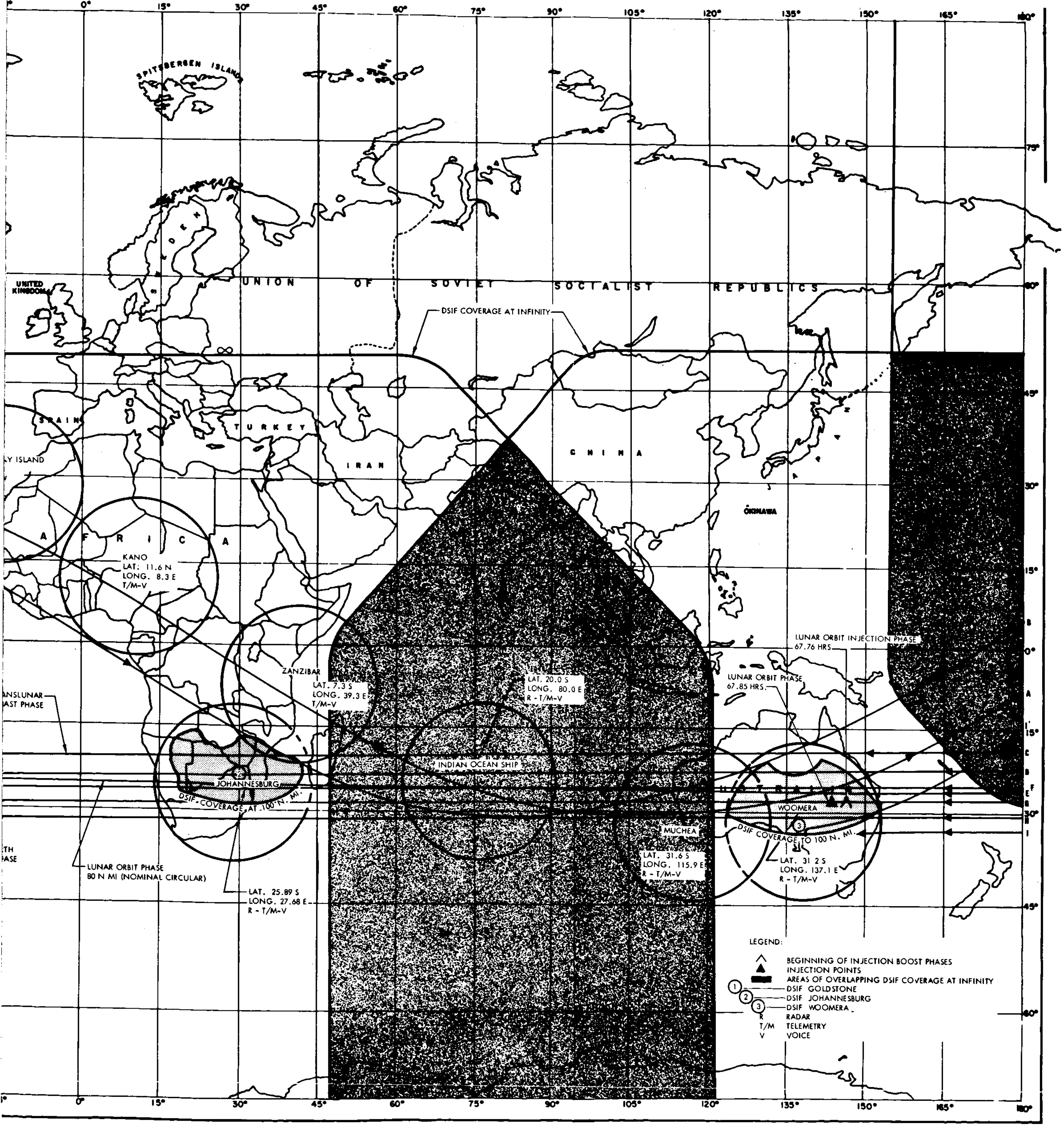

Figure 61. Mission Trajectory - GOSS Coverage

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TABLE 3

Goss Coverage Summary Lunar Landing Mission

ASCENT PHASE

Mission Time (Hrs.)	Phase Time (Sec.)	Station	Communications	Radar
0.0	0	Canaveral	In Contact	In Contact
0.02	72	Bermuda	Acquisition	
0.036	131	Bermuda		Acquisition
0.086	309	Canaveral		Loss
0.103	372	Canaveral	Loss	
0.196	706	Bermuda		Loss

EARTH PARKING ORBIT PHASE

Mission Time (Hrs.)	Phase Time (Min.)	Station	Communications	Radar
0.196	0.0	Bermuda	In Contact	Out of Contact
0.208	0.7	Bermuda	Loss	
0.276	4.8	Grand Canary	Acquisition	
0.296	6.0	Grand Canary		Acquisition
0.358	9.7	Grand Canary		Loss
0.371	10.5	Kano	Acquisition	
0.379	11.1	Grand Canary	Loss	
0.396	12.0	Kano		Acquisition
0.461	15.9	Kano		Loss
0.484	17.3	Kano	Loss	
0.549	21.2	Zanzibar	Acquisition	
0.568	22.3	Zanzibar		Acquisition
0.633	26.2	Zanzibar		Loss
0.661	27.9	Zanzibar	Loss	
0.709	30.8	Indian Ocean Ship	Acquisition	
0.724	31.7	Indian Ocean Ship		Acquisition
0.799	36.2	Indian Ocean Ship		Loss

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TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Min.)	Station	Communications	Radar
0.821	37.5	Indian Ocean Ship	Loss	
0.864	40.1	Muchea	Acquisition	
0.884	41.3	Muchea		Acquisition
0.949	44.6	Woomera (DSIF)*	Acquisition	
0.953	45.4	Woomera	Acquisition	
0.954	45.5	Muchea		Loss
0.959	45.8	Woomera (DSIF)*		Acquisition
0.963	46.0	Woomera		Acquisition
0.971	46.5	Muchea	Loss	
1.026	49.8	Woomera (DSIF)*		Loss
1.033	50.2	Woomera		Loss
1.044	50.9	Woomera (DSIF)*	Loss	
1.053	51.4	Woomera	Loss	
1.088	59.5	Canton	Acquisition	
1.209	60.8	Canton		Acquisition
1.263	64.0	Canton		Loss
1.286	65.4	Canton	Loss	
1.449	75.2	Point Arguello	Acquisition	
1.451	75.3	Guaymas	Acquisition	
1.459	75.8	Goldstone (DSIF)*	Acquisition	
1.464	76.1	Point Arguello		Acquisition
1.466	76.2	Guaymas		Acquisition
1.474	76.7	Goldstone (DSIF)*		Acquisition
1.489	77.6	Goldstone (DSIF)*		Loss
1.499	78.2	Houston	Acquisition	
1.506	78.6	Goldstone (DSIF)*	Loss	
1.508	78.7	Houston		Acquisition
1.509	78.8	Point Arguello		Loss
1.529	80.0	Point Arguello	Loss	

*It is assumed that tracking rates do not exceed DSIF capability.

[redacted]

TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Min.)	Station	Communications	Radar
1.536	80.4	Guaymas		Loss
1.548	81.1	Cape Canaveral	Acquisition	
1.556	81.6	Guaymas	Loss	
1.559	81.8	Cape Canaveral		Acquisition
1.584	83.3	Houston		Loss
1.593	83.8	Bermuda	Acquisition	
1.603	84.4	Houston	Loss	
1.606	84.6	Bermuda		Acquisition
1.629	86.0	Cape Canaveral		Loss
1.648	87.1	Cape Canaveral	Loss	
1.681	89.1	Bermuda		Loss
1.703	90.4	Bermuda	Loss	
2.109	114.8	Johannesburg	Acquisition	
2.129	116.0	Johannesburg		Acquisition
2.168	118.3	Johannesburg		Loss
2.179	119.4	Johannesburg	Loss	
2.263	124.0	Indian Ocean Ship	Acquisition	
2.283	125.2	Indian Ocean Ship		Acquisition
2.349	129.2	Indian Ocean Ship		Loss
2.364	130.1	Indian Ocean Ship	Loss	
2.409	132.8	Muchea	Acquisition	
2.429	134.0	Muchea		Acquisition
2.493	137.8	Woomera	Acquisition	
2.494	137.9	Muchea		Loss
2.509	138.8	Muchea	Loss	
2.511	138.9	Woomera		Acquisition
2.526	139.8	Woomera		Loss
2.539	140.6	Woomera	Loss	
2.741	152.7	Canton	Acquisition	
2.749	153.2	Canton		Acquisition

[redacted]

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TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Min.)	Station	Communications	Radar
2.759	153.8	Canton		Loss
2.763	154.0	Canton	Loss	
2.844	158.9	Kauai	Acquisition	
2.861	159.9	Kauai		Acquisition
2.896	162.0	Kauai		Loss

TRANSLUNAR INJECTION PHASE

Mission Time (Hrs.)	Phase Time (Min.)	Station	Communications	Radar
2.92	0.0	Kauai	In Contact	Out of Contact
2.93	25.7	Kauai	Loss	
2.96	159.6	Point Arguello	Acquisition	
2.97	189.5	Point Arguello		Acquisition
3.00	270.7	Goldstone (DSIF)	Acquisition	

TRANSLUNAR COAST PHASE

Mission Time (Hrs.)	Phase Time (Min.)	Station	Communications*	Radar*
3.01	0.0	Point Arguello	In Contact	In Contact
3.01	0.0	Goldstone (DSIF)	In Contact	In Contact
3.13	0.12	Point Arguello**	Loss	Loss
3.13	0.12	Goldstone (DSIF)	Loss	Loss
3.73	0.72	Johannesburg (DSIF)	Acquisition	Acquisition
7.03	4.02	Goldstone (DSIF)	Acquisition	Acquisition
7.81	4.80	Johannesburg (DSIF)	Loss	Loss

*Differences between Communications and Radar coverage are assumed to be negligible beyond "Earth Parking Orbit" phase.
**Assuming C-Band capability.

TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Min.)	Station	Communications*	Radar*
16.38	13.37	Woomera (DSIF)	Acquisition	Acquisition
17.79	14.78	Goldstone (DSIF)	Loss	Loss
19.71	16.70	Johannesburg (DSIF)	Acquisition	Acquisition
28.48	25.47	Woomera (DSIF)	Loss	Loss
31.28	28.27	Goldstone (DSIF)	Acquisition	Acquisition
31.66	28.65	Johannesburg (DSIF)	Loss	Loss
40.38	37.37	Woomera (DSIF)	Acquisition	Acquisition
41.81	38.80	Goldstone (DSIF)	Loss	Loss
43.71	40.70	Johannesburg (DSIF)	Acquisition	Acquisition
52.48	49.47	Woomera (DSIF)	Loss	Loss
55.51	52.50	Goldstone (DSIF)	Acquisition	Acquisition
55.66	53.65	Johannesburg (DSIF)	Loss	Loss
64.38	61.37	Woomera (DSIF)	Acquisition	Acquisition
65.41	62.40	Goldstone (DSIF)	Loss	Loss
67.59	64.58	Woomera (DSIF)	Loss***	Loss***

*Differences between Communications and Radar coverage are assumed to be negligible beyond "Earth Parking Orbit" phase.
**Assuming C-Band capability.
***S/C passes behind the moon.

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TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

TRANSLUNAR INJECTION PHASE

Mission Time (Hrs.)	Phase Time (Sec.)	Station	Communications	Radar
64.75	0	S/C Behind Moon	No Contact	No Contact
64.85	322			

LUNAR ORBIT PHASE - PRIOR TO LEM DEPARTURE

Mission Time (Hrs.)	Phase Time (Sec.)	Station (DSIF)	Communications	Radar
67.85	0.0	—	Out of Contact	Out of Contact
68.41	33.6	Woomera	Acquisition	Acquisition
69.46	96.6	Woomera	Loss	Loss
70.45*	156.0*	Johannesburg	Acquisition	Acquisition
70.45*	156.0*	Woomera	Acquisition	Acquisition
71.50*	219.0*	Johannesburg	Loss	Loss
71.50*	219.0*	Woomera	Loss	Loss
72.49*	278.4*	Johannesburg	Acquisition	Acquisition
72.49*	278.4*	Woomera	Acquisition	Acquisition

LUNAR ORBIT PHASE - FROM LEM SEPARATION TO DOCKING

Mission Time (Hrs.)	Phase Time (Sec.)	Station (DSIF)	Communications and Radar	Apollo-LEM Link
72.66	0.0	Johannesburg	In Contact	In Contact
72.66	0.0	Woomera	In Contact	
73.42*	45.6*	Johannesburg	Loss of LEM	
73.42*	45.6*	Woomera	Loss of LEM	
73.54*	52.8*	Johannesburg	Loss of Apollo	
73.54*	52.8*	Woomera	Loss of Apollo	
74.53*	112.2*	Johannesburg	Acquisition of Apollo	

*Parallax exists, thereby causing one DSIF Station to have an advantage in observing the disappearance of the S/C behind the moon, while the other Station has an advantage in observing the emergence from behind the moon. However, the difference is here assumed to be negligible.

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TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Sec.)	Station (DSIF)	Communications and Radar	Apollo-LEM Link
74.53*	112.2*	Woomera	Acquisition of Apollo	
74.65*	119.4*	Johannesburg	Acquisition of LEM	
74.65*	119.4*	Woomera	Loss of Apollo	
75.32	159.6			Loss
75.58	175.2	Johannesburg	Loss of Apollo	
76.57	234.6	Johannesburg	Acquisition of Apollo	
77.15	269.2			Acquisition
77.41	284.8			Loss
77.62	297.6	Johannesburg	Loss of Apollo	
78.61	357.0	Johannesburg	Acquisition of Apollo	
79.19	391.6			Acquisition
79.45	407.2			Loss
79.66	420.0	Johannesburg	Loss of Apollo	
80.65	479.4	Johannesburg	Acquisition of Apollo	
81.23	514.0	Goldstone	Acquisition of Apollo	
81.23	514.0			Acquisition
81.61	539.4	Goldstone	Loss of LEM	
81.61	539.4	Johannesburg	Loss of LEM	
81.70	542.4	Goldstone	Loss of Apollo	
81.70	542.4	Johannesburg	Loss of Apollo	
82.69	601.8	Goldstone	Acquisition of Apollo & LEM	

LUNAR ORBIT PHASE - SUBSEQUENT TO LEM DOCKING

Mission Time (Hrs.)	Phase Time (Min.)	Station (DSIF)	Communications	Radar
82.81	0.0	Goldstone	In Contact	In Contact
83.74	55.8	Goldstone	Loss	Loss

*Parallax exists, thereby causing one DSIF Station to have an advantage in observing the disappearance of the S/C behind the moon, while the other Station has an advantage in observing the emergence from behind the moon. However, the difference is here assumed to be negligible.

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TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Min.)	Station (DSIF)	Communications	Radar
84.73	115.2	Goldstone	Acquisition	Acquisition
85.78	178.2	Goldstone	Loss	Loss

TRANSEARTH INJECTION PHASE

Mission Time (Hrs.)	Phase Time (Sec.)	Station (DSIF)	Communications	Radar
86.20	0.0	S/C Behind Moon	No Contact	No Contact
86.24	127.5	S/C Behind Moon	No Contact	No Contact

TRANSEARTH COAST PHASE

Mission Time (Hrs.)	Phase Time (Hrs.)	Station	Communications	Radar
86.24	0.0	—	Out of Contact	Out of Contact
86.58	0.34	Goldstone (DSIF)	Acquisition	Acquisition
87.04	0.80	Woomera (DSIF)	Acquisition	Acquisition
90.94	4.70	Goldstone (DSIF)	Loss	Loss
94.34	8.10	Johannesburg	Acquisition	Acquisition
99.14	12.90	Woomera (DSIF)	Loss	Loss
105.94	19.70	Goldstone (DSIF)	Acquisition	Acquisition
106.34	20.10	Johannesburg (DSIF)	Loss	Loss
111.04	24.80	Woomera (DSIF)	Acquisition	Acquisition
114.54	28.30	Goldstone (DSIF)	Loss	Loss
118.29	32.05	Johannesburg (DSIF)	Acquisition	Acquisition

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TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Hrs.)	Station	Communications	Radar
123.14	36.90	Woomera (DSIF)	Loss	Loss
130.09	43.85	Goldstone (DSIF)	Acquisition	Acquisition
130.34	44.10	Johannesburg (DSIF)	Loss	Loss
135.04	48.80	Woomera (DSIF)	Acquisition	Acquisition
137.54	51.30	Goldstone (DSIF)	Loss	Loss
142.29	56.05	Johannesburg (DSIF)	Acquisition	Acquisition
147.14	60.90	Woomera (DSIF)	Loss	Loss
154.31	68.07	Goldstone (DSIF)	Acquisition	Acquisition
154.34	68.10	Johannesburg (DSIF)	Loss	Loss
159.04	72.80	Woomera (DSIF)	Acquisition	Acquisition
161.59	75.35	Goldstone (DSIF)	Loss	Loss
168.30	82.06	Canton	Acquisition	Acquisition
168.42	82.18	Woomera (DSIF)	Loss	Loss
168.50	82.26	Kauai	Acquisition	Acquisition
168.60	82.36	Canton	Loss	Loss
168.62	82.38	Kauai	Loss	Loss

ENTRY PHASE

Mission Time (Hrs.)	Phase Time (Sec.)	Station	Communications	Radar
168.65	0	—	Out of Contact	Out of Contact
168.80	540	Guaymas	Acquisition	Acquisition
168.83	660	White Sands	Acquisition	Acquisition

TABLE 3

Goss Coverage Summary Lunar Landing Mission (Cont'd)

Mission Time (Hrs.)	Phase Time (Sec.)	Station	Communications	Radar
168.87	708	San Antonio	Acquisition	Acquisition
168.87	709	Guaymas	Loss	Loss
168.87	714	White Sands	Loss	Loss

PARACHUTE DESCENT PHASE

Mission Time (Hrs.)	Phase Time (Sec.)	Station	Communications	Radar
168.90	0	San Antonio	In Contact	In Contact
169.04	509	San Antonio	Loss*	Loss*

*Earth landing is assumed to occur directly on target, i.e., in the immediate vicinity of the landing site radar.